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USAAEFA PROJECT NO. 80-17-2

AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST PART 2 YAH-64 ADVANCED ATTACK HELICOPTER

FINAL REPORT

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The Airworthiness and Flight Characteristics (A&FC) Evaluation Part 2, of the prototype YAH-64 helicopter (S/N 77-23258) was conducted at Palomar Airport, Carlsbad, California (elevation 328 ft). A total of 12 flights were conducted between 8 December and 17 December 1981 and 14.3 productive hours were flown. Prior to this test significant design changes were incorporated in the flight control system, the digital automatic stabilization equipment (DASE) and the stabilator system to correct objectionable characteristics determined during

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the airworthiness qualification program. Significant improvements in handling qualities were noted since the previous evaluation. Uncommanded control inputs, caused by recentering of the SAS actuators, upon failure or disengagement of the DASE may cause a potentailly hazardous situation. The instrument flight characteristics of the YAH-64 are satisfactory in smooth air but have yet to be evaluated in turbulent conditons. Manual programming of the stabilator in rearward flight did not significantly reduce objectional vibration at the pilot's station. Results of this test have shown that both previously reported deficiencies and 14 shortcomings have been corrected. One deficiency not previously observed, was identified: the possibility of a false indication of dual engine failure following a single engine failure. Two previously unreported shortcomings were also identified.

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SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 80-17-2, Airworthiness and Flight Characteristics Test, Part 2, YAH-64 Advanced Attack Helicopter

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1. The purpose of this letter is to establish the Directorate for Development and Qualification position on the subject report. The objectives of this test were to assess the handling characteristics, vertical climb performance, evaluate vibration characteristics in rearward flight, and to conduct a limited evaluation of instrument flight capability. Due to schedule restraints, the Airworthiness and Flight Characteristics (A&FC) test of the YAH-64 was divided into three parts, of which this is the second.

2. This Directorate agrees with the report conclusions and recommendations, with the exceptions identified herein. Dispositions of redesigned subsystems/components affecting the conclusions are also identified. Conclusions and recommendations are discussed by paragraph, as indicated.

a. Paragraph 45b. The uncommanded control inputs caused by recentering of the SAS actuators, though identified as a potentially hazardous condition, could not be adequately evaluated due to the limited scope of this program. Additionally, the final production configuration of the Digital Automatic Stabilization Equipment (DASE) software had not been determined prior to this evaluation. Further evaluation of DASE failures/disengagements must be conducted once the DASE software has been finalized for production. These additional tests are currently programmed to be conducted during Part 3 of this A&FC test program.

b. Paragraph 45c. The instrument flight characteristics, though satisfactory in smooth air, must be further evaluated in light to moderate turbulence. Further evaluation of the instrument flight capability of the YAH-64 is planned for A&FC, Part 3.

c. Paragraph 46. The engine out warning system will be changed prior to A&FC, Part 3, to activate at an engine power turbine speed (Np) of 89% rather than 94% as noted during this evaluation. Consideration should also be given to developing a comparator circuit which could sample an additional engine parameter (i.e. engine torque) to insure that an actual engine failure has occurred prior to activation of the system. The operation of the engine out warning system will be reevaluated during Part 3 of the A&FC test program.

DRDAV-D
SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 80-17-2, Airworthiness and Flight Characteristics Test, Part 2, YAH-64 Advanced Attack Helicopter

3. This report indicates that significant improvements in the handling qualities of the YAH-64 have been made since A&FC, Part 1. Additional testing will be required, however, to determine the instrument flight capability in light to moderate turbulence and to further evaluate the operation of the DASE once the production software configuration has been finalized. This additional testing is currently planned to be accomplished in conjunction with Part 3 of the A&FC test program.

FOR THE COMMANDER:

CHARLES C. CRAWFORD, JR.
Director of Development
and Qualification

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INTRODUCTION

BACKGROUND

1. In June 1973, the United States Army Aviation Systems Command, since renamed the US Army Aviation Research and Development Command (AVRADCOM), awarded a Phase I Advanced Development Contract to Hughes Helicopters Incorporated (HHI). The contract required HHI to design, develop, fabricate, and initiate a development/qualification effort on two Advanced Attack Helicopter prototypes and a ground test vehicle as part of a Government Competitive Test. The United States Army Aviation Engineering Flight Activity (USAAEFA) conducted Development Test I using two of these aircraft (ref 1, app A). In December 1976, AVRADCOM awarded a phase II Engineering Development Contract to HHI for further development and qualification of the YAH-64 to include subsystems and mission essential equipment. During this program, Engineer Design Tests (EDT) 1, 2 and 4 were conducted by USAAEFA to evaluate development progress (refs 2, 3 and 4). In December 1980 and January 1981 USAAEFA, in conjunction with US Army Aviation Development Test Activity, conducted EDT 5 to assess the integrated operation of all YAH-64 subsystems (ref 5). AVRADCOM requested USAAEFA to conduct this airworthiness and flight characteristics (A&FC) test following incorporation of changes to improve flying qualities, structural integrity, flight performance, and vibration characteristics of the prototype aircraft. Additional changes are planned for the production aircraft. Part 1 of this A&FC was completed in July 1981 (ref 6). USAAEFA was tasked in October 1981 to conduct Part 2 of this A&FC (ref 7). A test plan (ref 8) was submitted in October 1981, and an airworthiness release (ref 9) was issued in December 1981.

TEST OBJECTIVES

2. The objectives of this A&FC Part 2 test were to assess handling characteristics, vertical climb performance, evaluate vibration characteristics in rearward flight, and to conduct an evaluation of instrument flight capability. A follow-on Part 3 A&FC test, which will complete the A&FC phase of testing, is scheduled in May 1982.

DESCRIPTION

3. The YAH-64 is a two-place, tandem seat, twin-engine helicopter with four bladed main and antitorque rotors, conventional wheel landing gear and a movable horizontal stabilator. The helicopter is powered by two General Electric YT700-GE-700R

turboshaft engines rated at 1563 shaft horsepower (sea level standard day, uninstalled). A 30mm chain gun is mounted on the underside of the fuselage below the front cockpit. The helicopter has wings with two store pylons on each side to carry either HELLFIRE missiles, 2.75 inch folding fin aerial rockets, auxiliary fuel tanks or a combination thereof. An aerodynamic mockup of the Martin-Marietta Target Acquisition Designation System/Pilot Night Vision System (TADS/PNVS) was installed. The test helicopter was HHI air vehicle number 5 (US Army serial number 77-23258). Major changes to the helicopter since A&FC Part 1, as specified in reference 10, appendix A, are included in the aircraft description in appendix B. A more complete description of the test aircraft is also contained in the operator's manual (ref 11).

TEST SCOPE

4. Flight testing for A&FC, Part 2 was conducted at Palomar Airport, Carlsbad, California (elevation 328 ft). Twelve flights were conducted for a total of 14.3 productive flight hours. Tests were conducted during the period 8 through 17 December 1981. The original test plan called for a total of 35 productive flight hours with all tests to be conducted at Edwards Air Force Base, California (elevation 2300 ft) during the period 30 November 1981 through 9 January 1982. Due to changes in the YAH-64 program schedule, significant modification to the scope of this test were made. Priorities were established to insure that the maximum number of planned tests were accomplished during the specified time period. All tests were flown by a USAAEFA crew consisting of two test pilots. HHI installed, calibrated and maintained the test instrumentation and performed all aircraft maintenance during the test. Flight restrictions contained in the airworthiness release issued by AVRADCOM and the operator's manual were observed during this evaluation. Where possible, flight test data were compared with the system specification (ref 12) and results obtained during previous evaluations. The test conditions are shown in Table 1.

TEST METHODOLOGY

5. Established flight test techniques and data reduction procedures were used (refs 13 and 14, app A). Test methods are briefly discussed in the Results and Discussion section of this report. Flight test data were obtained from calibrated test instrumentation and were recorded on magnetic tape. Real time telemetry was used to monitor selected parameters throughout the

flight test. A detailed listing of the test instrumentation is contained in appendix C. Test techniques and data analysis methods are described in appendix D.

RESULTS AND DISCUSSION

GENERAL

6. Several design changes were incorporated in the flight control system, the Digital Automatic Stabilization Equipment (DASE), and the stabilator system. These changes were incorporated to correct objectionable characteristics determined during the airworthiness qualification program. Vertical climb performance testing was completed. Significant improvements in handling qualities were noted since Part 1 of this A&FC. The instrument flight characteritics of the YAH-64 were satisfactory in smooth air but have yet to be evaluated in turbulent conditions. Vibration characteristics were essentially unchanged from previous evaluations. Manual programming of the stabilator in rearward flight did not significantly reduce objectionable vibration at the pilot station. Both previously reported deficiencies and 14 shortcomings have been corrected. One deficiency, not previously observed, was identified: the possibility of a false indication of dual engine failure following a single engine failure. Two previously unreported shortcomings were also identified.

PERFORMANCE

Vertical Climb Performance

7. Vertical Climb Performance Tests were conducted at the conditions shown in Table 1. Two flights were accomplished at approximately the same gross weight and test conditions. The helicopter was initially stabilized in a hover at a 100-foot wheel height and then incremental power settings were used to climb vertically. Data were recorded during the stabilized free flight hover and again when the helicopter attained an unaccelerating vertical climb. The nondimensional test results for both the OGE hover data and vertical climb data are presented in figures 1 and 2 Appendix E.

8. The hover data presented in figure 1, Appendix E compared favorably with previous test results (Reference 4 and 6, Appendix A). The curve through the hover data was used as the basis for OGE hover power required to calculate the nondimensional vertical climb performance. This curve shows slightly more power required to hover than previous results with the helicopter similarly configured. Because of the extremely limited thrust coefficient range of the data, this curve was used only to determine the test day power required difference between hover and vertical climb.

Table 1. Test Conditions[1]

Type of Test	Gross Weight (lb)	Longitudinal Center of Gravity Location (FS)	Density Altitude (ft)	Trim Calibrated Airspeed Test (kt)	Remarks
Vertical Climb Performance	14,640	206.6 (AFT)	220	Zero	Wind less than 3 kts.
Control Positions[2] in Trimmed Forward	15,040	206.5 (AFT)	6580	42-112 43-112 38-108	Level Flight IRP[3] Climb Autorotation
Static Longitudinal Stability	14,880	206.0 (AFT)	5560	96	Level Flight
Maneuvering Stability	14,900	206.1 (AFT)	5860	100, 135	Steady turns, pull-ups, and pushovers
Dynamic Stability	14,640	206.0 (AFT)	5420	78, 100, 120	Longitudinal long period. and short period.
Controllability	14,980	206.7 (AFT)	5360	100	Cyclic only level flight, DASE ON & OFF
Instrument Flight[2,4] Capability	15,020	206.6 (AFT)	Field to 9500	0 to 120	Basic instrument flight and instrument flight profiles
DASE Evaluation	14,860	206.1 (AFT)	300	30 KTAS[5]	Low speed rearward flight
				0-86	Instrument takeoff, Hover, DASE failures
Vibration	14,740	206.5 (AFT)	200	0 to 40 KTAS	Rearward flight (Wind less than 5 kts).
				30 KTAS	Stabilator incidence from 35° LEU to -5° LEU Stabilator Fixed 35° LEU and -5° LEU
	15,500	206.4 (AFT)	5840	50 to 136	Level Flight

NOTES:

[1] 8-Hellfire configuration, 100% main rotor speed. All tests DASE ON and attitude HOLD OFF, except as otherwise noted

[2] DASE OFF

[3] IRP = Intermediate Rated Power

[4] DASE ON, Attitude HOLD ON and OFF.

[5] KTAS = Knots true airspeed

HANDLING QUALITIES

General

9. Tests were conducted to evaluate instrument flight capability and perform limited handling qualities tests, which included a verifiction of design changes. The test conditions are shown in table 1 and are representative of the conditions flown during Part 1 of this A&FC for those tests where comparative data was obtained. Both quantitative data and qualitative pilot comments were recorded during these tests. All tests were conducted in smooth air. The maximum surface wind during the tests was 8 knots.

Longitudinal Control System Characteristics

10. The longitudinal control system mechanical characteristics were evaluated on the ground with external hydraulic and electrical power applied to the aircraft and the rotors stopped. Force measurements were taken at the center of the pilot cyclic stick with the copilot/gunner cyclic stick extended and retracted. Tests were performed with trim feel ON and the cyclic position was retrimmed at 50 percent between forward and aft measurements. Separate measurements were taken using a digital force gauge to determine breakout force and Trim Feel System freeplay. Data are presented in figures 3 and 4, appendix E. Table 2 is a summary of the longitudinal control system mechanical characteristics observed during these tests. The longitudinal force gradient was increased from A&FC, Part 1 by modification of the force gradient spring. The breakout force was not reduced but is now more compatible with the higher force gradient and is satisfactory. The trim feel system freeplay of 0.3 inches has an adverse effect on trimmability. The longitudinal control system characteristics failed to meet the requirements of paragraph 10.3.2.1.1 of réference 12, appendix A in that the longitudinal breakout force (plus friction) exceeds the 1.5 pound limit by 0.9 pounds, however, longitudinal control characteristics are considered acceptable for an attack helicopter.

Control Positions in Trimmed Forward Flight

11. Control positions in trimmed forward flight were evaluated in level flight, climbs with intermediate rated power (IRP), and autorotation, in the 8-HELLFIRE configuration. Data are presented in figures 5 through 8, appendix E. Longitudinal control position gradients were conventional in level flight, IRP climbs, and autorotation throughout the speed range tested except between 40

Table 2. Longitudinal Control System Mechanical Characteristics[1]

Test Parameter	CONTROL SYSTEM	
	LONGITUDINAL[2]	LONGITUDINAL[3]
Breakout Force (plus friction) (lb)	2.4 lb fwd; 2.4 lb aft	2.3 lb fwd; 1.4 lb aft
Breakout Force (plus friction) (lb)	2.4 lb fwd; 2.4 lb aft	2.4 lb fwd; 1.2 lb aft
Full control travel (in.)	10.4	10.4
Trim displacement banc (in.)	0	0.15
Trim Feel System Freeplay	0.3	0.3
Limit control force (lb)	20.2 fwd; 18.6 aft	Not Evaluated
Control Centering	Positive	Positive
Control forces trimmable to zero	Yes	Yes
Force gradient (lb/in.)	1.3 fwd; 1.2 aft	1.9 fwd; 1.5 aft

NOTES:

[1]Rotor static ground hydraulic and electrical power applied
[2]Co-pilot gunner cyclic control stick retracted
[3]Co-pilot gunner cyclic control stick extended

to 60 knots calibrated airspeed (KCAS) where the gradient was approximately neutral. Trim changes with power variation were also determined at 100 KCAS. Between 30% and 100% of test day IRP engine power, longitudinal and lateral trim change was less than 0.2 in. All conditions were satisfactory.

Static Longitudinal Stability

12. The static longitudinal stability characteristics were evaluated at a trim speed of 96 KCAS in level flight by varying airspeed ±20 knots about trim in 5 knot increments. Data are presented in figure 9, appendix E. The longitudinal control position gradient was essentially neutral between approximately +3 and -8 knots from the trim airspeed. At airspeeds further from trim, up to ±20 KCAS, the longitudinal control position gradient was shallow. The shallow gradient over the forty knot airspeed band in combination with the 0.3 in. trim feel system freeplay and the longitudinal long term dynamic stability (para 15) resulted in poor trimmability over the airspeed range tested. The trimmability was further evaluated during simulated instrument meteorological conditions (IMC) (para 22). The neutral to shallow longitudinal control position variation with airspeed contibuted to moderate pilot workload to maintain trim in simulated IMC flight. A shallow longitudinal control position variation with airspeed is considered desirable for an attack helicopter and is satisfactory.

Maneuvering Stability

13. The maneuvering stability characteristics were evaluated using constant airspeed left and right steady turns and symmetrical pull-ups and pushovers. Data are presented in figures 10 through 13, appendix E. Maneuvering stability was significantly improved over A&FC, Part 1 as longitudinal control position and force gradients with load factor were essentially linear up to 2.7g, the maximum tested. The average force gradient was 3.0 lb/g. The previously reported shortcoming of a pitch "dig in" at 1.6g or greater was corrected and the maneuvering stability characteristics are satisfactory.

Dynamic Stability

14. The short-term dynamic stability characteristics were evaluated. Aircraft motions were induced by a single one inch longitudinal cyclic doublet at a frequency of one cycle per second. Following the input all controls were held fixed until the motion subsided. A typical time history of the short-term response is presented in figure 14, appendix E. The short-term

response was essentially deadbeat at all airspeeds tested and is satisfactory.

15. The longitudinal long-term dynamic stability characteristics were evaluated by displacing the longitudinal cyclic from trim and decreasing or increasing airspeed by 10 knots indicated airspeed (KIAS) then returning the cyclic to the trim position. All controls were then held fixed until recovery was initiated. Tests were conducted with DASE ON and attitude hold OFF. A typical time history of the long-term response is presented in figure 15, appendix E. The response at all airspeeds tested was a divergent oscillation with a period of approximately 90 seconds. The oscillation was controllable but, when added to the previously reported trim feel system freeplay (para 10) and weak longitudinal static stability (para 12), degraded trimmability. The longitudinal long-term response failed to meet the requirements of paragraph 10.3.4.2.1C of reference 12, appendix A in that the amplitude more than doubled in one oscillation, however, characteristics are considered acceptable for an attack helicopter.

Controllability

16. Controllability characteristics were evaluated in forward flight. Control step inputs were made, using a mechanical control fixture, in increasing increments to obtain the desired input. The inputs were held until a maximum rate was achieved or until recovery was necessary. All tests were conducted with attitude hold OFF and with DASE both ON and OFF. Data are presented in figures 16 through 21, appendix E.

17. The longitudinal control response, at 100 KCAS (fig. 16) was 11.5 degrees/sec/in. for forward and aft inputs DASE ON and OFF. Control sensitivity was greater with DASE ON (17 degrees/sec^2/in.) than DASE OFF (10 degrees/sec^2/in.) and the time to reach 63 percent of maximum pitch rate was 0.5 sec with DASE ON and 1.1 sec with DASE OFF. Aircraft response was rapid and no handling qualities problems were observed. The longitudinal controllability characteristics are satisfactory.

18. The lateral control response, at 100 KCAS (fig. 17) was 24 degrees/sec/in. for left and right inputs, DASE ON and OFF. Control sensitivity, DASE ON, was 60 degrees/sec^2/in. and 40 deg/sec^2/in. DASE OFF. Time to reach 63 percent of maximum pitch rate was 0.3 sec DASE ON and 0.7 sec DASE OFF. Though roll response was rapid, by comparison to other helicopters currently in the Army inventory, lateral controllability characteristics were considered satisfactory and desirable for an attack helicopter.

19. The control response with DASE ON, although the same as DASE OFF, both longitudinally and laterally, was much more desirable due to increased sensitivity. With DASE ON (figs. 18 and 19) maximum rate was reached quickly but then the rate damping provided by the DASE reduced this rate so that at the large roll attitudes achieved prior to recovery, the steady state roll rate was lower and comfortable. With DASE OFF (figs. 20 and 21), a maximum rate was not reached until just prior to recovery, giving a combination of a large attitude change and a high rate which was uncomfortable. The harmony between the longitudinal and lateral control response is satisfactory.

Instrument Flight Capability

20. A limited IMC evaluation was performed and consisted of a single flight conducted in smooth air. Maneuvers performed included climbs, descents, climbing and descending turns, tracking, simulated approaches and instrument take-offs. The normal mode of operation for IMC flight was with DASE ON and Attitude Hold ON. Operation with either DASE ON and Attitude Hold OFF or with DASE OFF was considered to be a degraded mode. IMC conditions were simulated by the installation of white curtains in the pilot station as shown in photos 1 and 2. All available flight instruments were used including the Electronic Attitude Direction Indicator. Turn rate indication was not available as both turn needles were inoperative. Navigation and instrument approaches were done using doppler way points as simulated Automatic Direction Finding stations. The flight instruments are shown in photo 3. The PNVS was not operational during this test. For this flight only, the ship's airspeed was connected to the pilot airspeed indicator. Maneuvers were flown DASE ON with the Attitude Hold both ON and OFF and with DASE OFF.

21. With the Attitude Hold engaged, airspeed was easy to maintain ±5 KIAS (HQRS 2) and attention could be diverted to other tasks such as scanning approach charts, setting radios, etc., without concern that airspeed would change. Turns could be flown with lateral cyclic only without concern for airspeed control or turn coordination. The cyclic could be released completely when the number of degrees to desired heading was one half the bank angle and the aircraft would roll wings level on heading. Climbs and descents could be initiated with hands off the cyclic. The only input required was slight directional control movement to counteract changes in engine torque. The instrument flight characteristics are satisfactory in smooth air with Attitude Hold engaged.

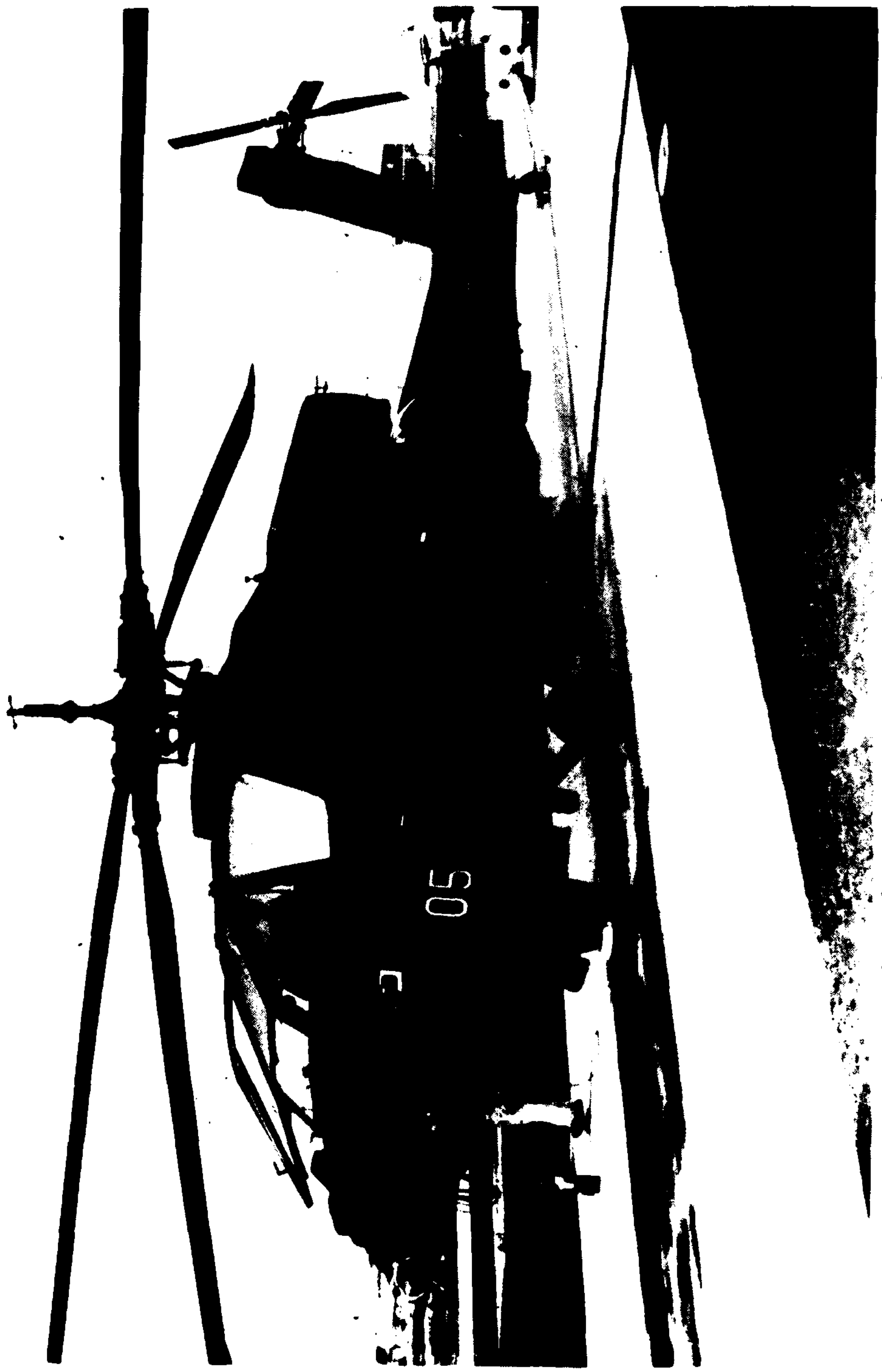

Photo 1. IMC Curtain Installation (exterior)

Photo 2. IMC Curtain Installation (interior)

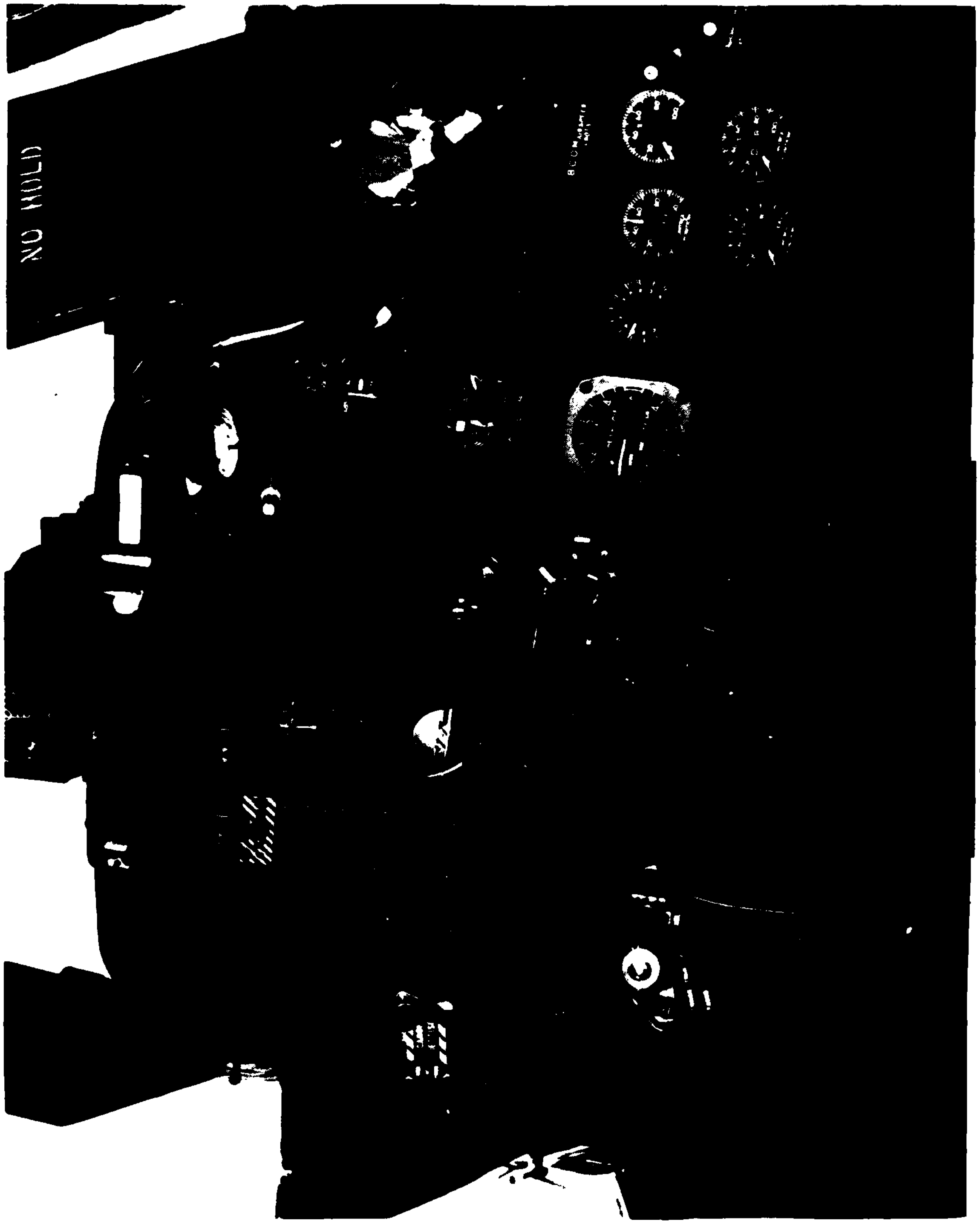

Photo 3. Pilot Instrument Panel

22. With the DASE ON and Attitude Hold OFF the previously mentioned problems of trim feel system freeplay (para 10), weak longitudinal static stability (para 12) and the divergent long-term dynamic stability (para 15) combine to make trimmability poor. The aircraft tended to wander nose low and the ability to maintain the trim airspeed ±10 knots required close monitoring by the pilot (HQRS 4). Aircraft control, however, was easily maintained and all approaches were successfully accomplished. Since the normal mode of operation for IMC flight is with DASE ON and Attitude Hold ON, trimmability is considered satisfactory for a degraded mode of operation during IMC flight in smooth air.

23. IMC characteristics were evaluated with the DASE OFF. Control characteristics of the aircraft were still adequate and typical maneuvers including a simulated ground controlled approach were flown with no significant problems. Maintaining trim airspeed within ±10 knots was difficult (HQRS 5-6) but is satisfactory for a degraded mode.

24. The IMC flight characteristics as tested during this limited evaluation were satisfactory in smooth air. Before final determination of IMC capability can be made, a more thorough evaluation should be conducted to include flight in light and moderate turbulence.

Aircraft Systems Failures

Generator Failures:

25. Simulated generator failures were performed by turning off either the No. 1 or No. 2 generator switch. A third DC electrical bus was installed in the aircraft to prevent system failures and erroneous warning activation with the failure of a single generator. Verification of the correct installation of the DC bus was performed by HHI personnel prior to the test. Initial tests caused erroneous activation of the engine out/low main rotor RPM audio tone. Maintenance action by HHI personnel corrected the problem and on subsequent tests no system failures or erroneous warnings occured. The one deficiency and two shortcomings, previously reported in A&FC, Part 1 (ref 6, app A) for generator failures, have been corrected and aircraft response to failure of a single generator is satisfactory.

Stabilator Failures:

26. Tests were conducted to determine if the stabilator system modifications listed in appendix B had corrected the problems

noted during A&FC, Part 1. Tests were performed to duplicate the flight conditions at which stabilator failures had previously occured. Initial tests resulted in spurious audio tones at 80 KIAS and failure of the stabilator to transition to the automatic mode when performing an acceleration in the manual mode. Maintenance action by HHI personnel, which included replacement of the Air Data System (ADS) processor, corrected the problem. Subsequent test results were satisfactory and the previously reported shortcomings associated with stabilator failures have been corrected.

Simulated Engine Failures:

27. Simulated single engine failures were performed both in flight and on the ground to evaluate the effectiveness of the engine out warning system. Failures were simulated in flight by retarding one engine power lever to the idle position. Failures were simulated on the ground by turning the main fuel switch off. Ground tests were performed with the engine power levers in the FLY and IDLE positions, and in the electrical control unit LOCKOUT mode. Prior to conducting the tests, system installation was verified by HHI personnel as being in accordance with the appropriate blueprint drawing.

28. Tests performed in flight showed that the power turbine speed (N_P) engine out warning system could give an erroneous indication of an engine failure. This erroneous failure, indicated by activation of the engine out warning light on the operating engine, occurred whenever main rotor speed (N_R) was allowed to droop to approximately 93 to 94 percent. This reduction in N_R was accompanied by a corresponding reduction in N_P which activated the engine out warning. This condition occurred repeatedly when one power lever was retarded to the idle position at a flight condition which then caused the fully operational engine to be limited by turbine gas temperature (TGT). An actual single engine failure under similiar flight conditions would cause illumination of both engine out warning lights even though only one engine had failed. This situation would be particularly hazardous when hovering OGE or during low speed terrain flight where a pilot may interpret a single engine failure as a dual engine failure, enter autorotation and land immediately when, in fact, it may have been possible to transition to single engine flight. The possibility of a false indication of dual engine failure following a single engine failure is a deficiency which should be corrected prior to testing with the YT 700-GE-701 engines installed.

29. Tests performed on the ground showed that no engine out warning was provided if the engine power levers were in other

than the fly position. The system should activate in response to an engine gas generator speed (N_g) of 63 percent or less, however, it would not operate in accordance with maintenance test procedures performed by HHI personnel. The engine out warning system was not in accordance with design requirements and as configured is a shortcoming.

Digital Automatic Stabilization Equipment Evaluation

30. Tests were conducted to determine if DASE modifications listed in appendix B were effective in correcting previously reported deficiencies and shortcomings. Test conditions flown were representative of those during A&FC, Part 1.

31. A low voltage monitor was incorporated which appeared to have corrected the previously reported deficiency of yaw Stability Augmentation System (SAS) hardovers to 20 percent of main flight control servoactuator travel. Flight conditions where previous failures had occurred were duplicated and no yaw hardover failures were experienced. The persistent yaw oscillations of ± 2 to ± 4 degrees, previously reported as a shortcoming in A&FC, Part 1 were not apparent during this evaluation.

32. The Hover Augmentation System (HAS) was evaluated in a stabilized hover, a bob up, and while repositioning with and without retrimming. No abrupt aircraft responses were noted, however, divergent pitch and roll oscillations were observed (fig. 22, app E). The period of both oscillations was approximately 11 seconds. The pilot was required to make continuous longitudinal control inputs, in order to damp the oscillations and could not safely remove his hand from the cyclic. The divergent pitch and roll oscillations, while hovering with the HAS engaged, is a degradation from previous evaluations, and is a shortcoming. Since all tests were conducted in winds of 8 knots or less, the HAS should also be evaluated in gusty wind conditions.

33. Single and multiple channel DASE failures were performed during flight with Attitude Hold ON and OFF and with HAS ON and OFF. Single channel failures were simulated by using the yaw, roll and pitch switches located on the DASE control panel. Multiple channel failures were simulated by depressing the DASE release switch on the pilot cyclic grip. It was found that a DASE failure resulted in uncommanded control inputs in all three axes because the SAS actuators were displaced from center. When the DASE failed (simulated by depressing the trim release switch) the SAS actuators immediately recentered causing an abrupt

uncommanded control input (fig. 23, app E). This condition is similar in abruptness and potentially of the same magnitude as an unmonitored hardover condition where the SAS actuator rapidly displaces from center, to either of its limits, and remains there. Previous evaluations included only monitored hardover tests, which were simulated electronically by rapidly displacing the SAS actuator from center, to either of its limits, and immediately returning the actuator to center. The monitored hardover test simulated the function of the DASE hardover monitor circuit (designed to prevent a hardover condition) but did not evaluate the abrupt recentering of an offset SAS actuator. Since the SAS actuators may also be displaced from center during HAS OFF operations (para 34), uncommanded control inputs may be a significant problem during NOE flight, confined area operations, operation in turbulence, IMC flight, flight using the PNVS as well as flight with the HAS engaged. Data obtained during controllability tests (para 16) show that uncommanded control inputs could produce rates of up to 25 degrees per second in pitch and 21 degrees per second in roll with an additional input in yaw. These rates would exceed the 10 degree per second maximum rate allowed by the system specification. Additionally, in the present configuration, it is possible to induce a condition equivalent to a three-axis hardover, with the failure of the Heading and Attitude Reference System (HARS). The magnitude of the hardover condition would be dependent upon the amount of SAS actuator displacement from center. Uncommanded control inputs caused by recentering of the SAS actuators, upon failure or disengagement of the DASE, may cause a potentially hazardous situation. Conditions capable of producing abrupt uncommanded control inputs should be further investigated and, if necessary, the DASE should be modified prior to reevaluation during A&FC, part 3.

34. Roll Command Augmentation System (CAS) washout times were reduced from 20 to 10 seconds in an attempt to correct the previously reported shortcoming of reduced rate damping authority caused by excessive CAS and SAS washout times (ref 6, app A). Tests were conducted to duplicate those previously reported and results show essentially no change (fig. 24, app E). The roll SAS actuator position is offset approximately 35 to 40 percent of its authority from the hover trim position which is shown in figure 25. The condition was not confined to the roll channel. This is substantiated by the time history of an instrument takeoff showing the pitch SAS actuator offset resulting in reduced rate damping authority in the forward (nose down) direction (fig 26). Additionally, the offset SAS actuator position could result in the equivalent of an unmonitored hardover caused by recentering of the SAS actuator following a DASE failure as discussed in

paragraph 33. The offset SAS actuator position did not produce undesirable handling qualities during this evaluation (all tests conducted in smooth air), however, the reduction of rate damping authority, a previously reported shortcoming, may be significant in turbulence. Further evaluation of the DASE should be conducted in light and moderate turbulence.

VIBRATION CHARACTERISTICS

General

35. The vibration characteristics of the YAH-64 were evaluated with the vertical vibration absorber removed. All tests were conducted in the 8-Hellfire configuration at a nominal CG location of 206.5 (aft). While the aft CG is more representative of the mission configuration, previous tests have shown vibration levels to be higher with a forward CG location (refs 4 and 6, app A). Qualitative evaluations were made at both crew stations. Quantitative data were gathered at the conditions shown in table 1.

Forward Flight

36. The vibration characteristics in level flight were essentially the same as those previously reported for similar flight conditions and aircraft configuration (ref 4, app A). Data are presented in figures 27 through 31, appendix E. The 4/rev (19.3 Hz) vertical vibration was not objectionable at either crew station between the range of 50 to 130 KCAS and vibration levels were slight (VRS 2-3).

Rearward Flight

37. The vibration characteristics were evaluated in rearward flight from hover to 40 knots true airspeed (KTAS) in 5 knot increments, at stabilator incidence angles of 35 degrees leading edge up (LEU) and 5 degrees leading edge down (LED). Tests were also conducted in rearward flight at 30 KTAS while varying stabilator incidence angles from 35 degrees LEU to 5 degrees LED in 5 degree increments. Data are presented in figures 32 through 46, appendix E. The 4/rev (19.3 Hz) vertical vibration was objectionable (VRS 4-6) at the pilot station at airspeeds greater than 15 KTAS (figs. 32 and 37). Vibration levels were greater at the 5 degree LED position than the 35 degree LEU position. The highest vibration levels (0.5g) were at 30 KTAS with stabilator incidence angles of 15 to 25 degrees LEU. Manual programming of the stabilator produced a slight reduction in 4/rev vertical vibration at 5, 10, and 35 degrees LEU (fig 42 and

43), however, the vibration remained objectionable at the pilot station (VRS 5). Throughout the test, vibration levels at the copilot station were not objectionable. Efforts to reduce the objectionable 4/rev vertical vibration at the pilot station should continue.

RELIABILITY AND MAINTAINABILITY

38. The reliability and maintainability features of the YAH-64 aircraft were evaluated throughout the test. Twelve equipment performance reports (EPR) were prepared and submitted during this evaluation and are listed in appendix F. This section is intended to summarize the most significant reliability and maintainability problems encountered.

39. Throughout the evaluation, HARS/DASE interface problems were encountered. The HARS would not give proper alignment indications. The DASE could be engaged during the alignment of the HARS and with the Heading (HDG) warning flag visible in the Horizontal Situation Indicator. The HARS unit was replaced just prior to A&FC, Part 2, and was replaced twice during the 8-day evaluation. Three EPR's associated with the HARS/DASE interface problem were submitted during the course of this evaluation. The HARS/DASE interface problem should be investigated and its impact on system reliability be determined.

40. Throughout the test program, accumulation of oil was observed on the deck below the main transmission and inside each engine cowling. Additionally, oil could be seen dripping from the tail boom as it had accumulated in the maintenance access area aft of the main transmission. Oil accumulation on the TGT electrical connection was shown to have resulted in fluctuating TGT indications (EPR No. 80-17-2-09). The majority of the oil appeared to be leaking from the No. 1 engine nose gearbox. A total of 26 ounces of oil was used during a 7 day period (8.9 flight hours) as indicated in EPR No. 80-17-2-12. The excessive oil accumulation due to leakage of the engine nose gearbox is a previously reported shortcoming.

MISCELLANEOUS

41. Testing was conducted to evaluate deficiencies and shortcomings identified during previous tests for which HHI personnel indicated corrective action had been taken, either as procedural changes or modifications to the test aircraft (S/N 77-23258). Those areas where corrective action had not been specified by HHI were not reevaluated during this program.

42. The following previously reported deficiencies have been corrected.

a. YAW SAS hardover failures (20 percent of main flight control servo actuator travel).

b. Disengagement of the DASE following failure of the No. 2 generator.

43. The following previously reported shortcomings have been corrected.

a. Illumination of the master caution light without the illumination of a caution, warning or advisory panel segment light.

b. Failure of the APU to start.

c. Erroneous activation of the engine out/main rotor RPM audio warning tone with failure of the No. 1 or No. 2 generator.

d. Erroneous activation of the stabilator audio warning tone with failure of the No. 1 generator.

e. Persistent yaw oscillations of ± 2 to ± 4 degrees in level flight.

f. Failure of the stabilator to consistently achieve complete automatic programming from a full LEU manual setting (35 deg LEU) when accelerating through the stabilator switching velocity.

g. The absence of SAS pitch rate damping at load factors greater than 1.6 due to saturation of the pitch SAS actuator.

h. The excessive longitudinal breakout force (specification noncompliance).

i. The inadvertent directional control inputs during brake application.

j. Failure of the stabilator automatic mode of operation while in a low power descent.

k. The lack of an acceptable method of sampling the primary and utility hydraulic fluid.

l. The annoying tone present in the intercom system.

m. The illumination of the master caution light with green advisory segment lights.

n. The failure of the fire bottle discharge lights to illuminate with the activation of the PRESS TO TEST switch.

AIRSPEED CALIBRATION

44. Airspeed calibration tests were conducted using the trailing bomb method. Tests were performed in level flight, climb, descent and autorotation. Calibrations were conducted on both the right and left pitot-static airspeed systems, and the ADS longitudinal airspeed. Airspeeds were calibrated from 40 to 120 KCAS. Data are presented in figures 47 through 56, appendix E. Figure 56 is a comparison of airspeed position errors during level flight, climb, and autorotation for pilot (right), copilot/gunner (left), and ADS systems. The ADS was also calibrated in low speed forward and rearward flight using a ground pace vehicle (para 5, app D). Though airspeed errors were greatest in an IRP climb, the position error did not appear to cause a problem during IMC flight and is considered satisfactory.

CONCLUSIONS

GENERAL

45. Based on the A&FC, Part 2 flight test of the YAH-64 helicopter, the following conclusions were reached:

a. Significant improvements in handling qualities have been made since A&FC, Part 1 (paras 13, 25, 26 and 31).

b. Uncommanded control inputs caused by recentering of the SAS actuators, upon failure or disengagement of the DASE, may cause a potentially hazardous situation (para 33).

c. The instrument flight characteristics are satisfactory in smooth air (para 24).

d. Manual programming of the stabilator during rearward flight, did not reduce vibration to an acceptable level at the pilots station (para 37).

e. Disengagement of the DASE following failure of the No. 2 generator, a previously reported deficiency, has been corrected (para 25).

f. Yaw SAS hardover failures, a previously reported deficiency has been corrected (para 31).

g. Fourteen previously reported shortcomings have been corrected (para 43).

h. Twelve equipment performance reports were submitted during this evalution (para 38).

i. One previously unreported deficiency has been identified.

j. Two previously unreported shortcomings have been identified.

DEFICIENCY

46. The following deficiency was identified (see app D for definition of deficiency used in this report):

The possibility of a false indication of dual engine failure following a single engine failure (para 28).

SHORTCOMINGS

47. The following shortcomings were identified (see app D for definition of shortcomings used in this report):

a. The engine out warning system not in accordance with design requirements (para 29).

b. The divergent pitch and roll oscillations while hovering with the HAS engaged (para 32).

SPECIFICATION COMPLIANCE

48. The YAH-64 was found to be in noncompliance with the following paragraphs of the Phase II Advanced Attack Helicopter System Specification AMC-SS-AAH-H10000A. Additional specification noncompliance, beyond the scope of the evaluation may exist.

*a 10.3.2.1.1 longitudinal breakout force (plus friction) in excess of 1.5 lb (para 10)

*b. 10.3.4.2.1c long term dynamic response amplitude more than doubled in one oscillation (para 15)

* Considered acceptable

RECOMMENDATIONS

49. The following recommendations are made:

a. Correct the deficiency in paragraph 46 prior to evaluation with the YT700-GE-701 engines installed.

b. Correct the shortcomings in paragraph 47 prior to A&FC, Part 3 testing.

c. Investigate conditions capable of producing abrupt uncommanded control inputs and if necessary the DASE should be modified prior to reevaluation during A&FC, Part 3 (para 33).

d. Conduct further testing of the instrument flight capability in light and moderate turbulence (para 24).

e. Conduct further evaluation of the DASE to include operation of the Hover Augumentation System (paras 32 and 34).

f. Continue efforts to reduce objectionable 4/rev vertical vibration at the pilot station (para 37).

g. Investigate the HARS/DASE interface problem and determine its impact on system reliability (para 39).

APPENDIX A. REFERENCES

1. Final Report, USAAEFA Project 74-07-2, *Development Test 1, Advanced Attack Helicopter Competitive Evaluation, Hughes YAH-64 Helicopter*, December 1976.

2. Final Report, USAAEFA Project No. 77-36, *Engineer Design Test 1, Hughes YAH-64 Advanced Attack Helicopter*, September 1978.

3. Final Report, USAAEFA Project No. 78-23, *Engineer Design Test 2, Hughes YAH-64, Advanced Attack Helicopter*, June 1979.

4. Final Report, USAAEFA Project No. 80-03, *Engineer Design Test 4, YAH-64 Advanced Attack Helicopter*, January 1980.

5. Final Report, AVNDTA, *Engineer Design Test, Government (EDT-G) 5 of the Advanced Attack Helicopter (AAH) (YAH-64)*, May 1981, revised June 1981.

6. Final Report, USAAEFA Project No. 80-17-1, *Airworthiness and Flight Characteristics Test Part 1 YAH-64 Advanced Attack Helicopter*, September 1981.

7. Letter, AVRADCOM, DRDAV-DI 21 October 1981, subject: Airworthiness and Flight Characteristics (A&FC) Test of the YAH-64 Advanced Attack Helicopter, Prototype Qualification Test-Government (PQT-G), Part 2 - Revision 1.

8. Test Plan, USAAEFA Project No. 80-17-2, *Airworthiness and Flight Characteristics Test, Part 2, YAH-64 Advanced Attack Helicopter*, October 1981.

9. Letter, AVRADCOM, DRDAV-D, 4 December 1981, subject: Airworthiness Release for Airworthiness and Flight Characteristics Test Part 2 of YAH-64 Helicopter, S/N 77-23258.

10. Letter, Hughes Helicopters, D. McGettigan, 4 December 1981, subject: AVO-5 Aircraft Configuration changes since POT Part 1.

11. Draft Technical Manual, TM 53-1520-238-10, *Operator's Manual for Army YAH-64 Helicopter*, 1 May 1981, with Change 1, 10 August 1981.

12. Specification, Hughes Helicopters, AMC-SS-AAH-H10000A, "YAH-64, Phase II Advanced Attack Helicopter Systems," 10 December 1976.

13. Pamphlet, USAMC, AMCP 706-204, "Engineering Design Handbook Helicopter Performance Testing," August 1974.

14. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Helicopter Stability and Control*, 10 June 1968.

APPENDIX B. DESCRIPTION

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GENERAL

1. The YAH-64 Advanced Attack Helicopter (fig. 1) is a tandem, two-place twin turbine engine, single main rotor aircraft manufactured by Hughes Helicopters Incorporated (HHI). The main rotor is a four-bladed fully articulated system supported by a stationary mast which transmits flight loads directly to the fuselage. The tail rotor is a four-bladed semi-rigid, delta-hinged system incorporating elastromeric teetering bearings. The rotors are driven by two General Electric YT 700-GE-700R engines through the power train shown in figure 2. An AiResearch GTCP 36-55(c) auxiliary power unit (APU) is installed primarily for starting the engines and to provide electrical and hydraulic power when the aircraft is on the ground and rotors are not turning. The aircraft is designed to carry various combinations of ordinance stores internally in the ammunition bay and externally on the four wing store positions. The YAH-64 is designed to operate during day, night and marginal weather combat conditions using the Martin-Marietta Target Acquisition Designation System (TADS)/Pilot's Night Vision System (PNVS). The test aircraft, S/N 77-23258, (photos 1 through 5) was configured with an aerodynamic mockup of the TADS/PNVS, 30mm chain gun, and HELLFIRE missile launcher loaded with four dummy missiles on each of the two inboard wing pylons. The major modifications and external configuration changes since A&FC, Part 1 are presented in table 1. The Back Up Control System (BUCS) was not operational during this test.

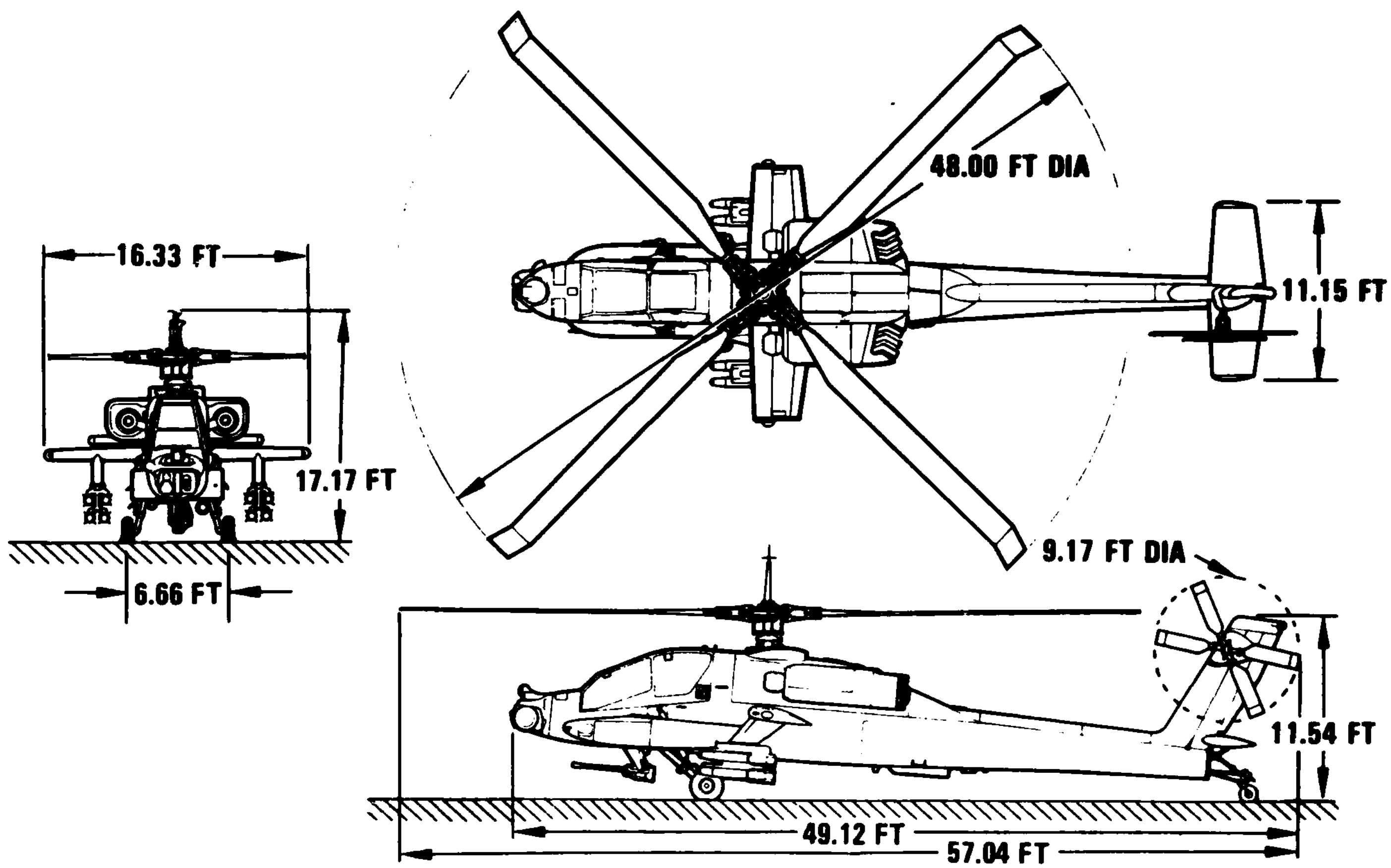


Figure 1. Aircraft Dimensions

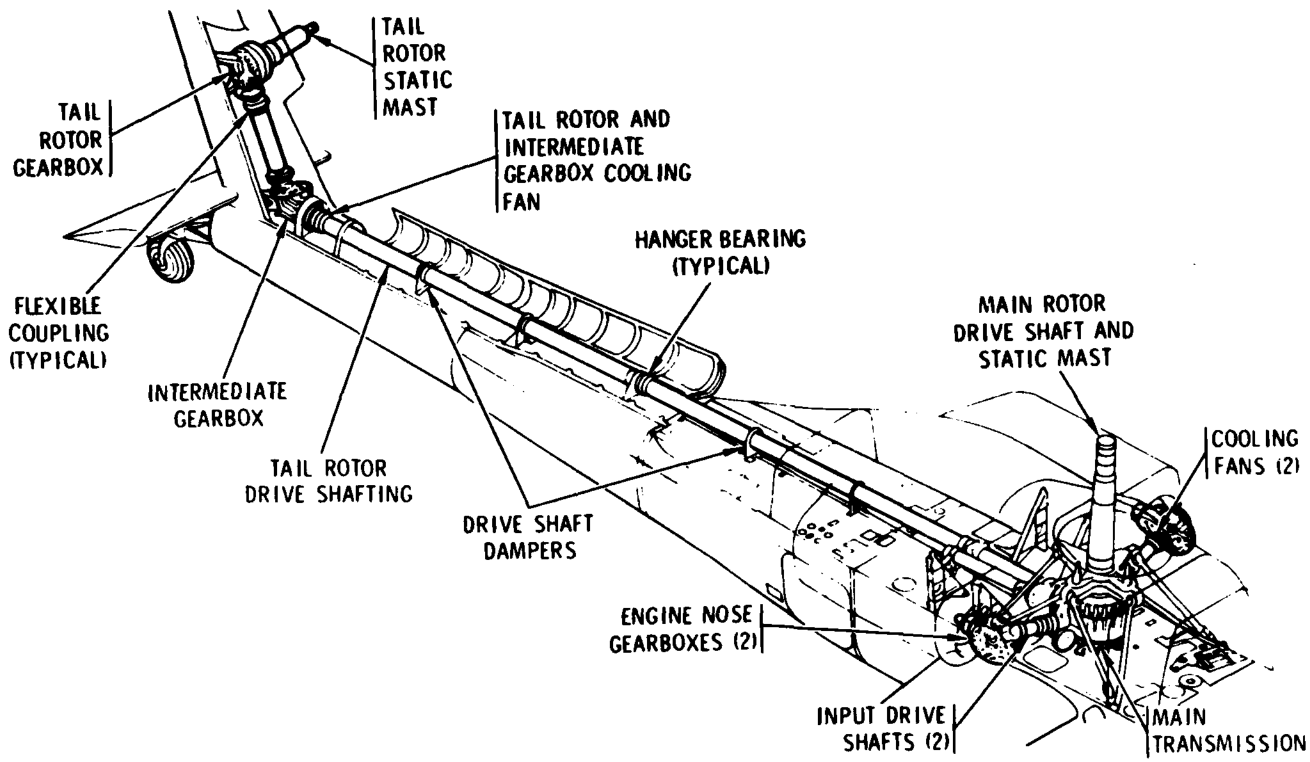


Figure 2. Powertrain

Photo 1. Left View

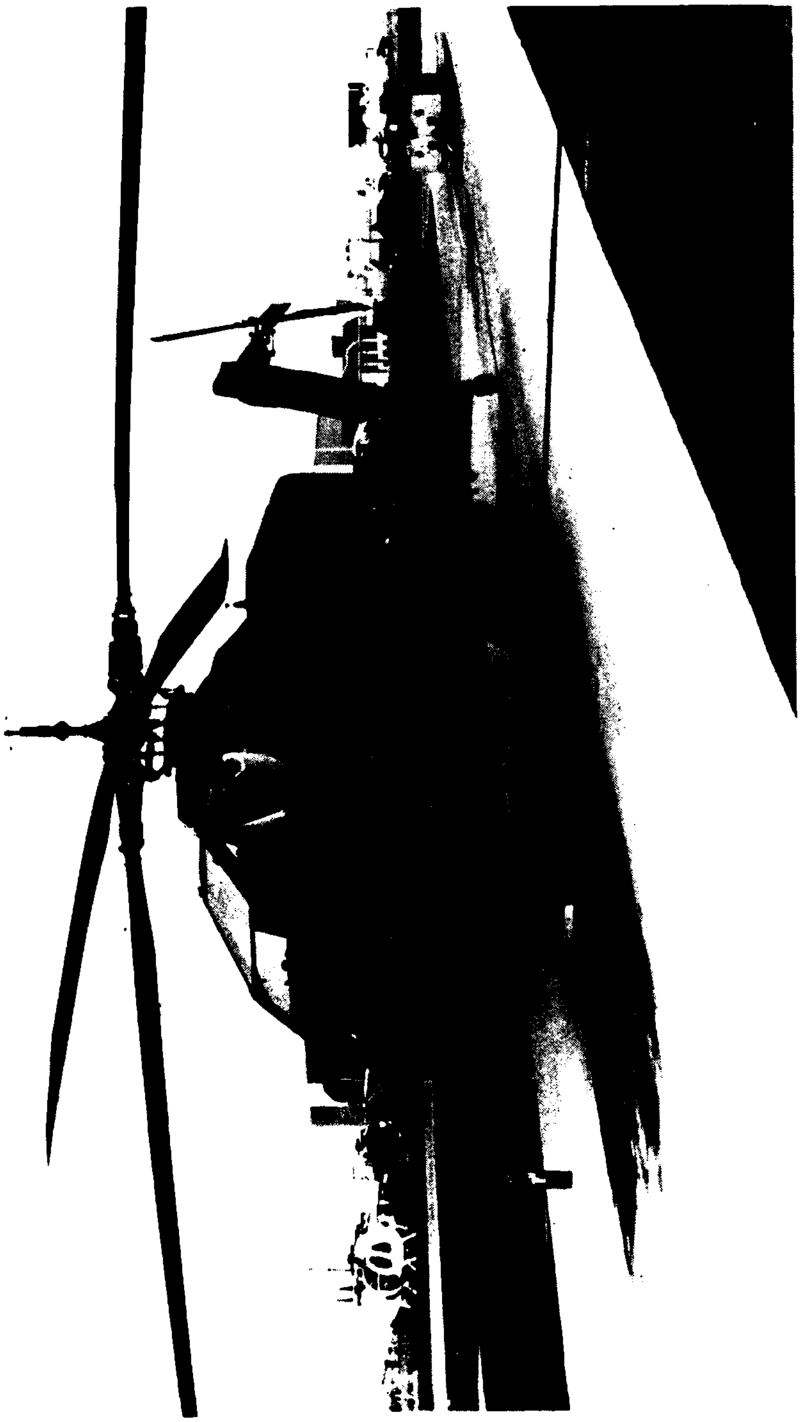


Photo 2. Left Front Quartering View

Photo 3. Front View

Photo 4. Right Front Quartering View

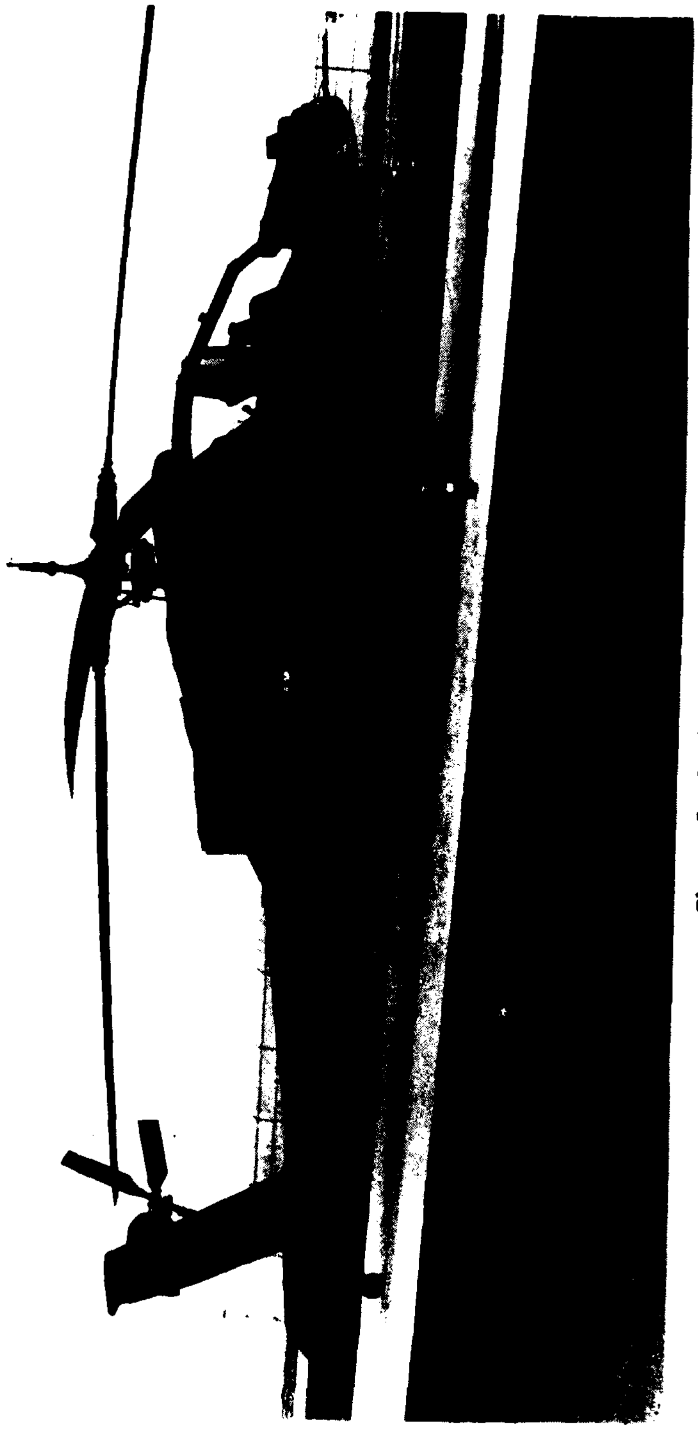


Photo 5. Right View

Table 1. Aircraft Configuration Changes
Since YAH-64 A&FC Part 1 (AV05)

1. Installed wing tip light fairings.

2. Replaced #1 instrumented M/R blade with a standard non-instrumented blade.

3. Removed M/R instrumentation rotating Pulse Code Modulation Cannister.

4. Replaced instrumented M/R HUB Assy. New unit has elastromeric feather bearings.

5. Replaced instrumented M/R pitch change link.

6. Replaced #2 engine with a calibrated engine.

7. Replaced both G07 engine ECU'S with G04HB units.

8. Replaced longitudinal control actuator with a modified unit that allows 20% FWD and 10% AFT SAS authority.

9. Modified the mechanical longitudinal control force gradient.

10. Cyclic discrete trim OFF switch disabled.

11. M/R pitch housing lead-lag link stops adjusted to conform to design requirements.

12. APU fuel control replaced with a high-rate unit.

13. Installed Telephonics ICS control boxes.

14. Installed amber caution lights for APU ON and BUC ON.

DIMENSIONS AND GENERAL DATA

Main Rotor

Diameter (ft)	48
Blade chord (in.)	21.0*
Main rotor total blade area (ft^2)	166.5
Main rotor disc area (ft^2)	1809.56
Main rotor solidity (thrust weighted, no tip loss)	0.092

*Includes tips

Airfoil	HH-02**
Twist	-9 deg
Number of blades	4
Rotor speed at 100 percent N_R (RPM)	289.3
Tip speed at 100 percent N_R (ft/sec)	727.09

Tail Rotor

Diameter (ft)	9.17
Chord; constant (in.)	10
Tail rotor total blade area (ft^2)	14.89
Tail rotor disc area (ft^2)	66.0
Tail rotor solidity	0.2256
Airfoil	NACA 632-414 (modified)
Twist (deg)	8.8 washout
Number of blades	4
Rotor speed at 100 percent N_R (RPM)	1403.4
Distance from main rotor mast centerline (C_L) (ft)	29.67
Tip speed at 100 percent N_R (ft/sec)	673
Teetering angle (deg)	35
Maximum blade angle (deg)	27

Horizontal Stabilator

Weight (lb)	77.3
Area (ft^2)	33.36
Span (ft)	11.15
Tip chord (ft)	2.65
Root chord (ft)	3.60
Airfoil	NACA 0018
Geometric aspect ratio	3.41
Incidence of chord line (deg)	Variable (45 degrees leading edge up to 10 degrees leading edge down).
Sweep of leading edge (deg)	2.89
Sweep of trailing edge (deg)	-7.23 deg
Dihedral (deg)	0

**Outer 20 inches swept 20 degrees and transitioned to an NACA 006 airfoil

Vertical Stabilizer

Area (from boom C_L) (ft ^{2})	32.2
Span (from boom C_L) (in)	113.0
Root chord (at boom C_L) (in)	44.0
Geometric aspect ratio	2.5
Airfoil	NACA 4415 (modified)
Sweep of Leading edge (deg)	29.4
Vertical stabilizer trailing edge deflection	16 deg left above W.L. 196.0

Wing

Span (ft)	16.33
Mean aerodynamic chord (in.)	45.9
Total area (ft^2)	61.59
Flap area (ft^2)	8.71 (fixed)
Airfoil at root	NACA 4418

Aircraft

Fuel quantity (gals.)	369
Design gross weight	14660
Maximum gross weight (lbs)	17850

FLIGHT CONTROL DESCRIPTION

General

2. The YAH-64 helicopter employs a single hydromechanical irreversible flight control system. The hydromechanical system is mechanically activated with conventional cyclic, collective and directional pedal controls, through a series of push-pull tubes attached to four airframe mounted hydraulic servoactuators. The four hydraulic servoactuators control longitudinal cyclic, lateral cyclic, main rotor collective and tail rotor collective pitch. Cyclic and directional servoactuators incorporate integral SAS actuators. Hydraulic power is supplied by two independent 3000-psi hydraulic systems which are powered by hydraulic pumps mounted on the accessory gearbox of the main transmission to allow full operation under a dual-engine failure condition. A Digital Automatic Stabilization Equipment (DASE) system is installed to provide rate damping and command augmentation. The DASE is limited to ± 10 percent of control authority in roll and yaw. A new longitudinal cyclic hydraulic servoactutor installed since A&FC, Part 1 allows 20 percent forward and 10 percent aft control authority in the pitch axis.

The DASE also provides attitude hold and a Hover Augmentation System (HAS). An electrically-actuated horizontal stabilator is attached to the lower aft portion of the vertical stabilizer. Movement of the stabilator can be controlled either manually or automatically. A trim feel system is incorporated in the cyclic and directional controls to provide a control force gradient with control displacement from a selected trim position. A trim release switch, located on the cyclic grip, provides momentary interruption of the trim feel system in all axes simultaneously to allow the cyclic or pedal controls to be placed in a new reference trim position. A modification of the trim feel system since A&FC, Part 1 removed the discrete off feature. Full control travel is 10.4 inches in the longitudinal control, 8.4 inches in the lateral control, 12.6 inches in the collective control, and 5.0 inches in the directional pedals.

Cyclic Control System

3. The cyclic control system (fig. 3) consists of dual-tandem cyclic sticks attached to individual support assemblies in each cockpit. The support assembly houses the primary longitudinal and lateral control stops, and two linear variable differential transformers (LVDT) designed to measure electrically the longitudinal and lateral motions of the cyclic for DASE computer inputs. A series of push-pull tubes and bellcranks transmit the motion of the cyclic control to the servoactuators and the mixer assembly. Motion of the mixer assembly positions the nonrotating swashplate, which transmits the control inputs to the rotating swashplate to control the main rotor blades in cyclic pitch (fig. 4). The cyclic stick grips are shown in figure 5. A stick fold linkage is provided to allow the copilot/gunner (CPG) to lower the cyclic stick to prevent interference when operating the weapon systems.

Collective Control System

4. The collective pitch control system (fig. 6) consists of dual-tandem collective sticks which transmit collective inputs to the main rotor through a series of push-pull tubes and bellcranks attached to the collective servoactuator. Motion of the servoactuator is transmitted through the mixer assembly to the swashplate to control the main rotor blades in collective pitch. Collective inputs are also transmitted to the load demand spindle of each engine hydromechanical unit (HMU). The HMU meters the fuel as appropriate to provide collective pitch compensation. Located at each collective control base assembly are the primary control stop, LVDT, and a one g balance spring. The LVDT supplies electrical inputs to the stabilator control units.

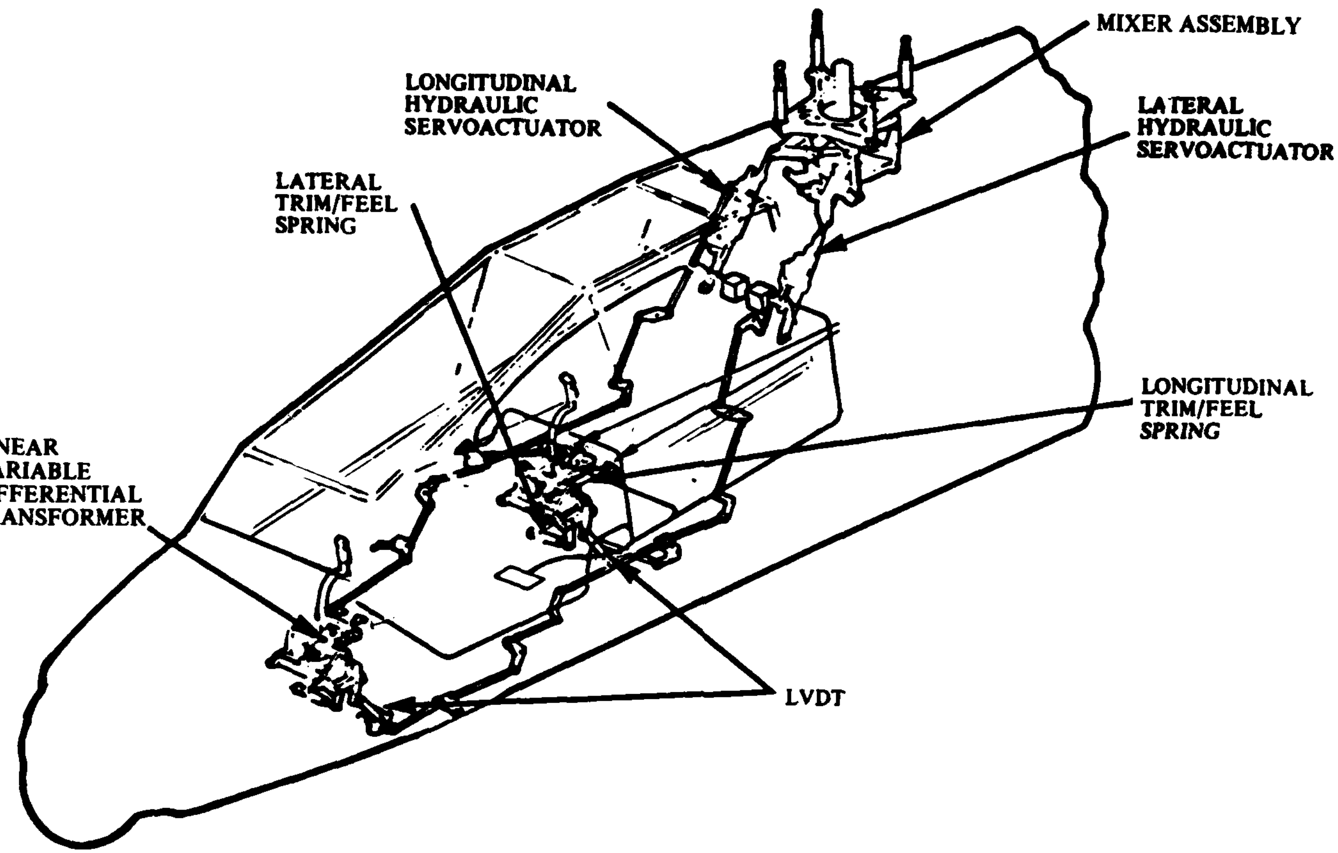


Figure 3. Cyclic Control System

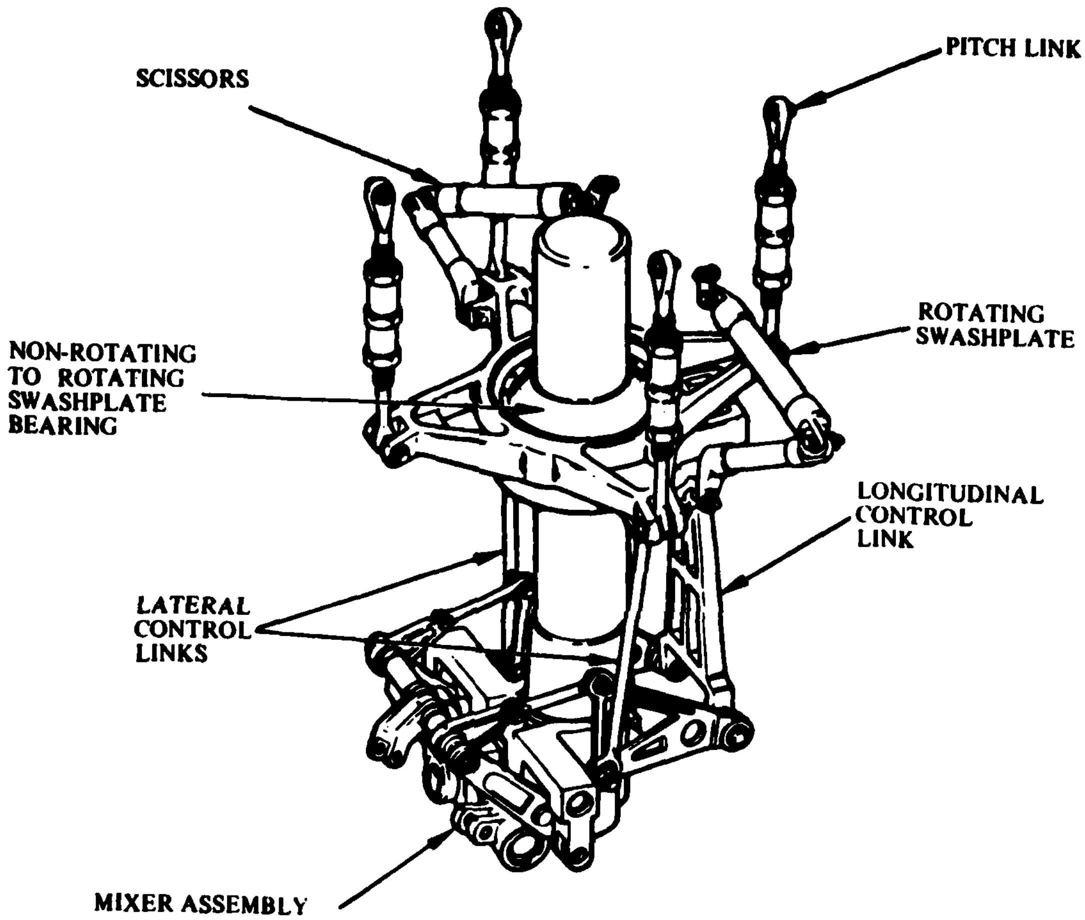
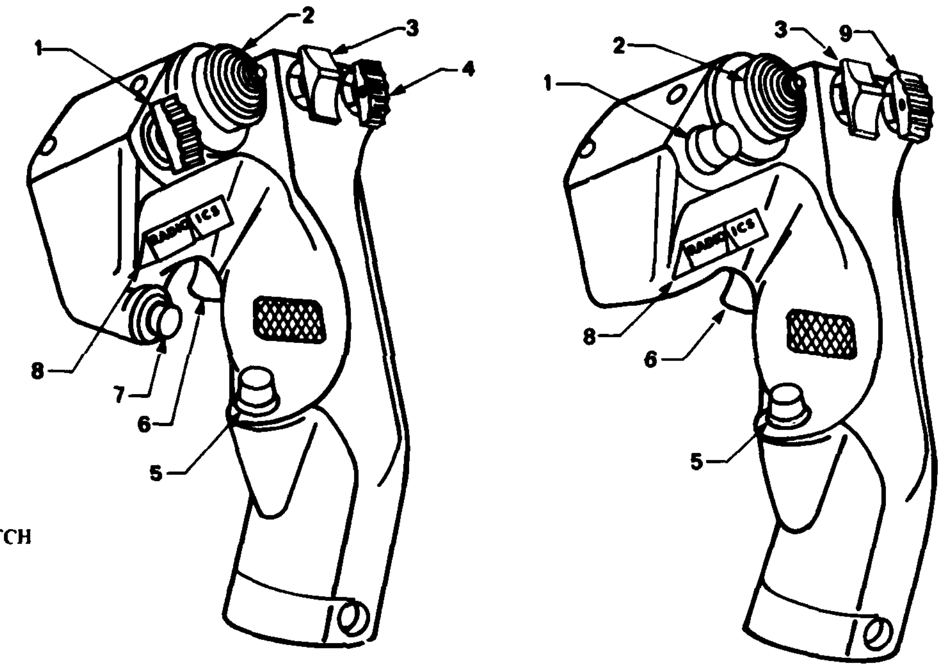


Figure 4. Main Rotor Swashplate Assembly

PILOT CYCLIC STICK GRIP

COPILOT/GUNNER CYCLIC STICK GRIP



1 TRIM RELEASE SWITCH (MOMENTARY ONLY)

2 WEAPONS ACTION SWITCH

3 FLIGHT MODE SYMBOLOGY SWITCH

4 TO BE DETERMINED

5 DASE RELEASE SWITCH

6 GUARDED TRIGGER SWITCH

7 REMOTE TRANSMITTER SELECTOR SWITCH (PILOT GRIP ONLY)

8 RADIO, ICS ROCKER SWITCH

9 NEAR/FAR FOCUS

Figure 5. Cyclic Stick Grips

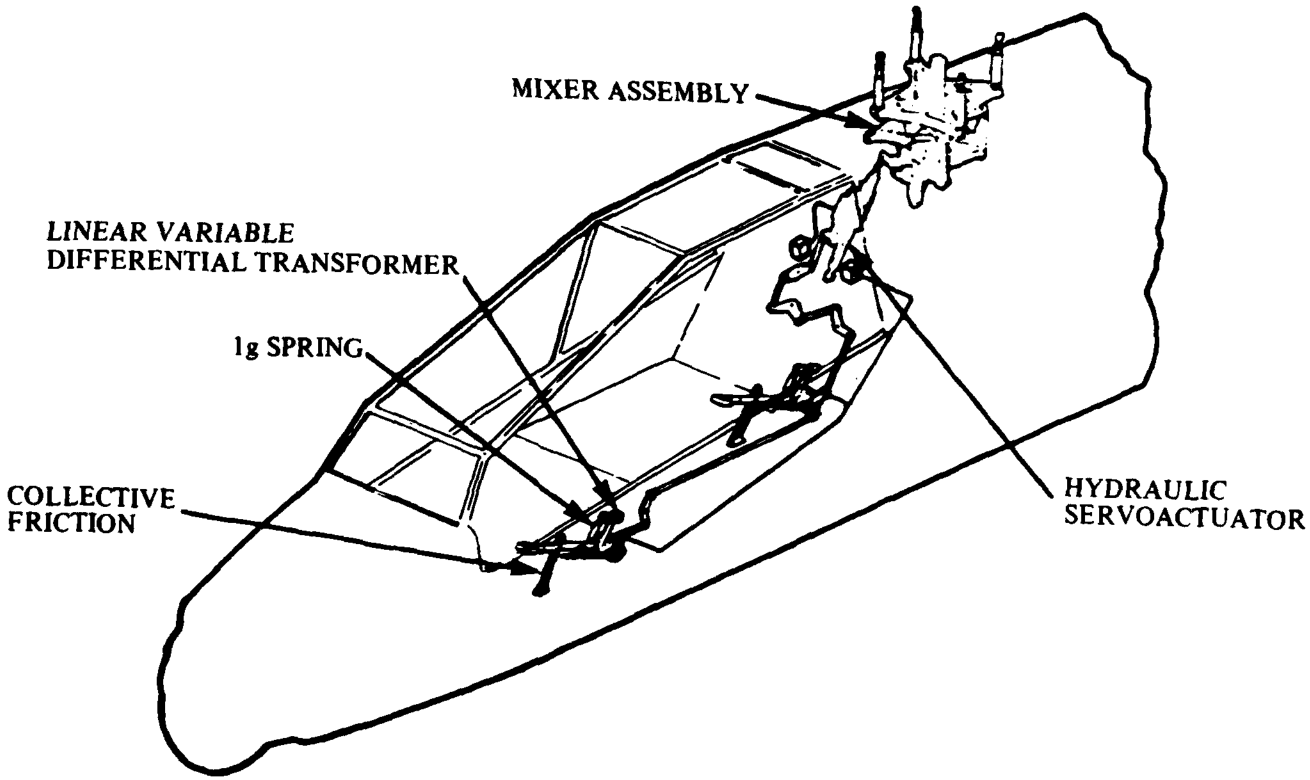


Figure 6. Collective Control System

5. Each collective stick (fig. 7) incorporates a switch box assembly, an engine chop collar, a stabilator control panel and an adjustable friction control. The engine chop collar allows rapid deceleration of both engines to flight idle, primarily to allow immediate action in the event of a tail rotor failure.

Directional Control System

6. The directional control system (fig. 8) consists of a series of push-pull tubes and bellcranks which transmit directional pedal inputs to the tail rotor hydraulic servoactuator located in the vertical stabilizer. Attached to each directional pedal assembly is a primary tail rotor control stop and one LVDT. Two sets of wheel brake cylinders are attached to the directional pedals and a 360 degree swiveling tail wheel is incorporated. The tail wheel may be locked in the trailing position by means of a switch located on the pilot instrument panel.

Trim Feel System

7. A trim feel system is incorporated in the longitudinal, lateral and directional control systems. The trim feel system uses individual magnetic brake clutch assemblies in each of the control linkages. Trim feel springs are incorporated to provide a control force gradient and positive control centering. The electromagnetic brake clutch is powered by 28 VDC and is protected by the TRIM circuit breaker. A complete DC electrical failure will disable the trim feel system and allow the cyclic and directional pedals to move freely without resistance from the trim feel springs. The trim release switch on the pilot and CPG grip allows momentary release of the trim feel system.

Horizontal Stabilator

8. The horizontal stabilator is attached to the lower aft portion of the vertical stabilizer. A dual, series 28 VDC electromechanical actuator allows incidence changes of +45 to-10 degrees leading edge up (LEU) of travel. Safety features include an automatic shutdown capability which allows operation in the manual mode by means of a stabilator control panel located on each collective stick. An audio tone is associated with the failure of the automatic mode of operation. A stabilator kill switch, located on the pilot collective stick, disables both the automatic and manual modes operation to protect against a hardover failure (S/N 77-23258 only). There are three modes of stabilator operation: the automatic mode, the NOE/Approach mode and the manual mode. The stabilator is controlled in the automatic mode by two stabilator control units (SCU's). Each SCU controls one side of the dual actuator. Both SCU's receive collective

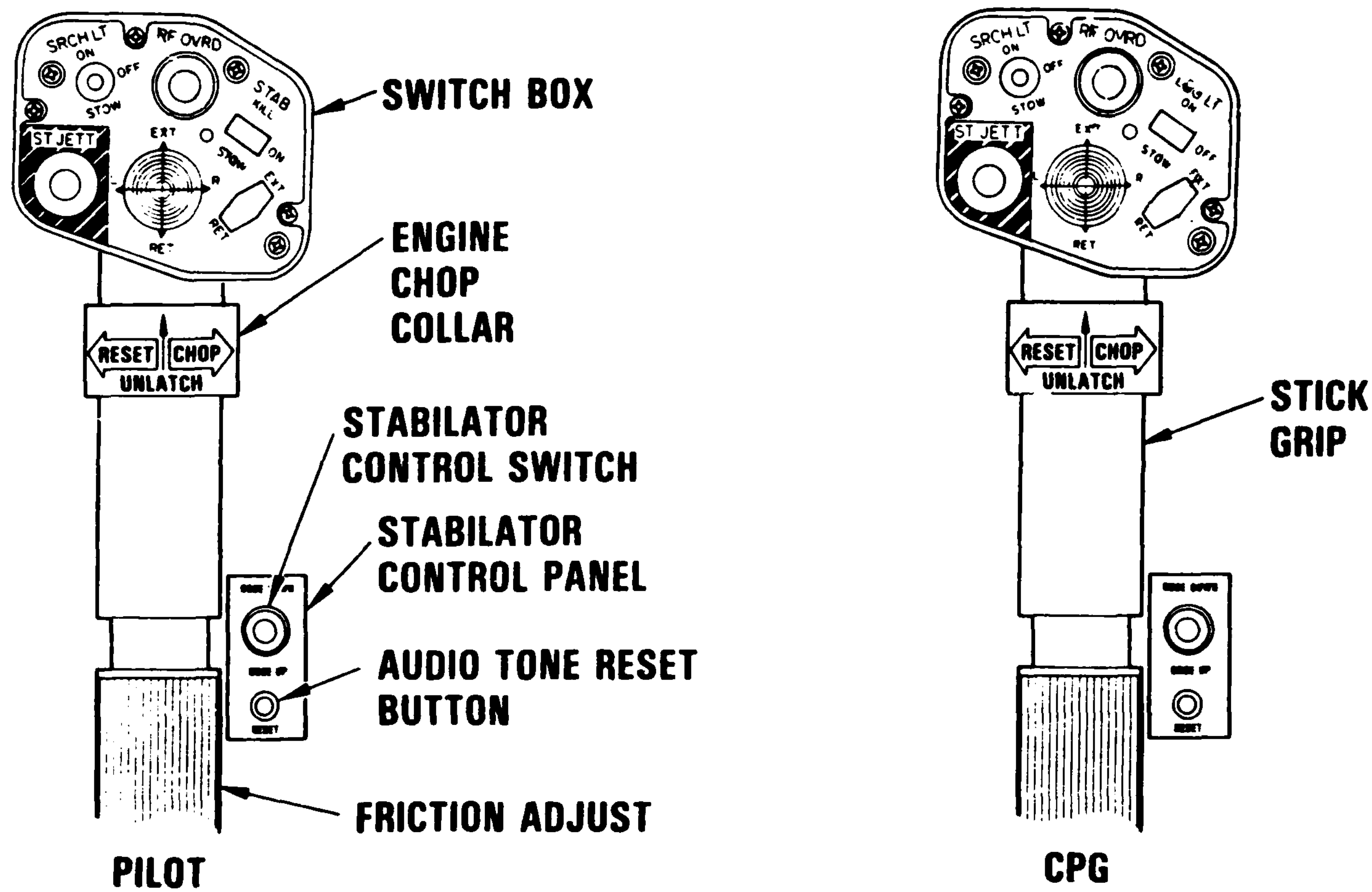


Figure 7. Collective Stick and Switch Box

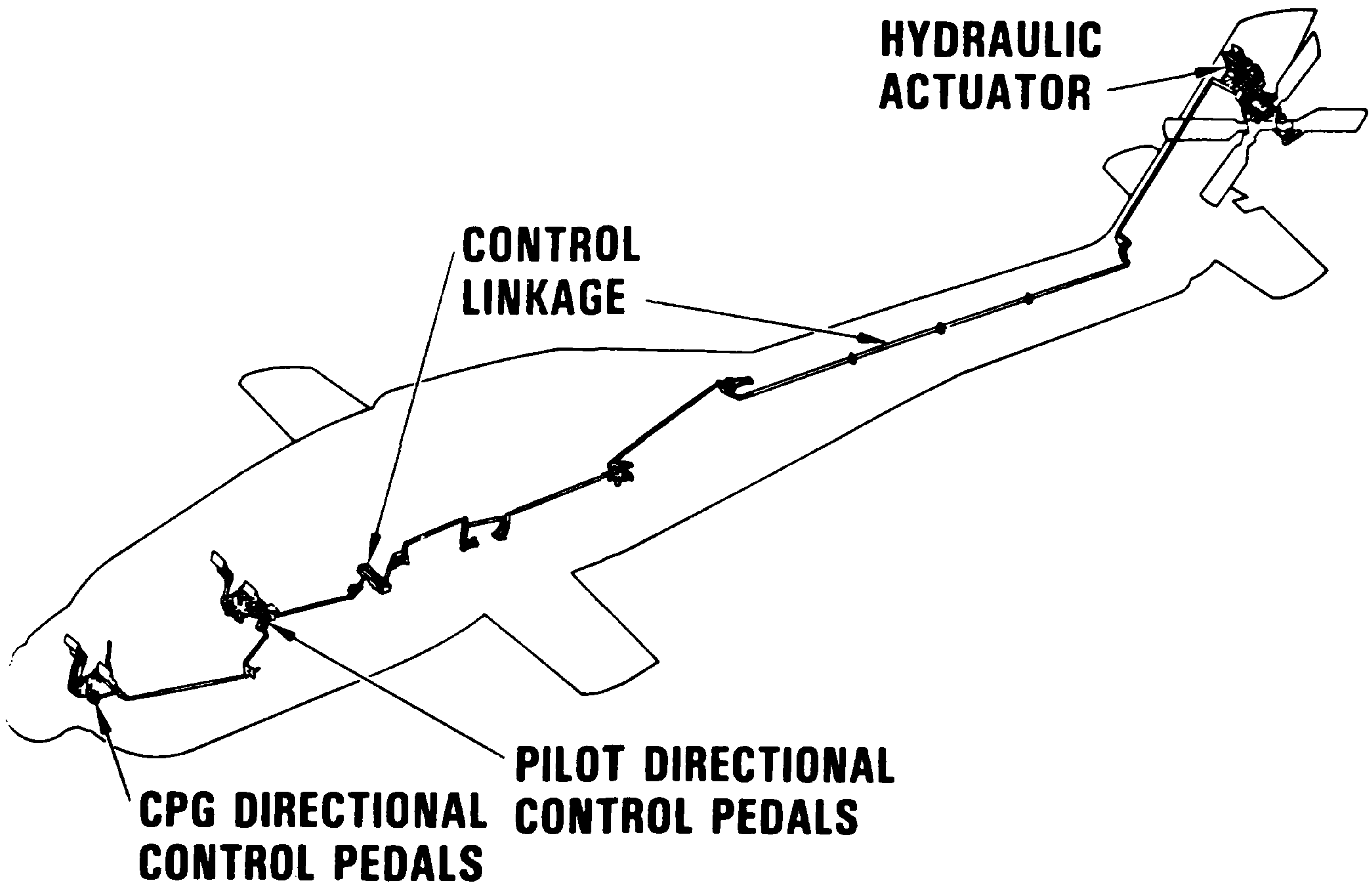


Figure 8. Directional Controls

control position information from redundant LVDT's. Two independent pitch rate gyros provide pitch rate information to the SCU's (one gyro for each SCU). The Air Data System (ADS) provides airspeed to both SCU's. Additionally, the left-hand pitot-static system supplies airspeed to one SCU and the right-hand system provides airspeed to the other SCU. Both SCU's receive position information from both sides of the dual actuator. The maximum rate of stabilator travel is 7 degrees per second. Modifications to the horizontal stabilator system since A&FC, Part 1 are presented in Table 2.

Table 2. Stabilator System Modifications Since YAH-64 A&FC 1 (AV05)

1. Installation of a filter to reduce electromagnetic interference (EMI) which have caused spurious stabilator fail audio warnings.

2. Movement of the auto stabilator DC circuit breakers to the 3rd bus.

3. Modification of program monitor #1 to change threshold from 25° LEU to 30° LEU.

4. Modification of airspeed bias from 40 to 58 KTS.

5. Inclusion of 0.3 sec time lag in collective and rate stabilator control paths.

6. Deactivation of current and rate monitors.

9. The automatic mode is operational when the aircraft has normal AC and DC electrical power applied. Automatic positioning of the stabilator during flight is primarily a function of airspeed and collective position. The stabilator also responds with a low gain (0.2 deg/sec/sec) and limited authority ($\pm$5.0 deg) to pitch rate inputs to the SCU. Software in the SCU limits the incidence change in the automatic mode from +25 to -5 deg LEU.

10. The NOE/Approach Mode is selected through the NOE/APPR mode switch on the pilot DASE panel and will stay engaged at any speed. The mode becomes operational below 80 knots indicated airspeed (KIAS) and will bias the stabilator to 25 degrees LEU at a 3.6 deg/sec rate. The mode can be disengaged by manual mode selection below 80 knots, activation of the DASE release, or by the AUTO STAB reset switch. Acceleration through 80 KIAS in the NOE/ approach mode will engage the automatic schedule and the stabilator will move at a 3.6 deg/sec rate for 10 seconds

or until the commanded position is reached. The stabilator then reverts to normal automatic scheduling of 7 deg/sec. Failure to revert to the automatic schedule will result in system disengagement with both visual and aural indications.

11. The manual mode can be selected below 80 KIAS through the pilot and CPG manual control switch on either collective stick. Manual control selection will result in STAB FAIL caution light illumination. Selection of automatic mode can be accomplished by pressing the AUTO STAB reset switch on the pilot or CPG collective stick. The stabilator will move at a 3.6 deg/sec rate for 10 seconds or until the automatic mode schedule position is reached. Acceleration through 80 KIAS in manual mode will engage the normal automatic mode and the stabilator will move at a 7 deg/sec rate.

12. The SCU's have a fault detection feature which will switch the stabilator mode of operation from automatic to manual if any of the following conditions are sensed:

a. A mismatch between the positions of the two sides of the actuator equivalent to 10 degrees of stabilator travel (if there is a runway failure of one side of the actuator, this feature will disable the automatic mode after 10 degrees of stabilator movement.)

b. The stabilator at a position of 20 degrees or greater with an airspeed greater than 110 KIAS.

c. The stabilator at a position of 30 degrees or greater with an airspeed greater than 80 KIAS (for 1 second or longer).

d. AC power to the collective position LVDT's less than 23 VAC.

<u>Flight Control Rigging</u>

13. A flight control rigging check was performed in accordance with procedures outlined in HHI Experimental Test Procedure (ETP) 7-211500000, dated 1 December 1980, (main and tail rotor controls) and ETP 7-211123600, dated 21 April 1980 (horizontal stabilator). The horizontal stabilator schedule is shown in figure 9. Tables 3 and 4 present the collective and cyclic rigging. Tail rotor rigging is shown below:

Full right pedal: 15.1 degrees thrust to left
Full left pedal: 27.0 degrees thrust to right

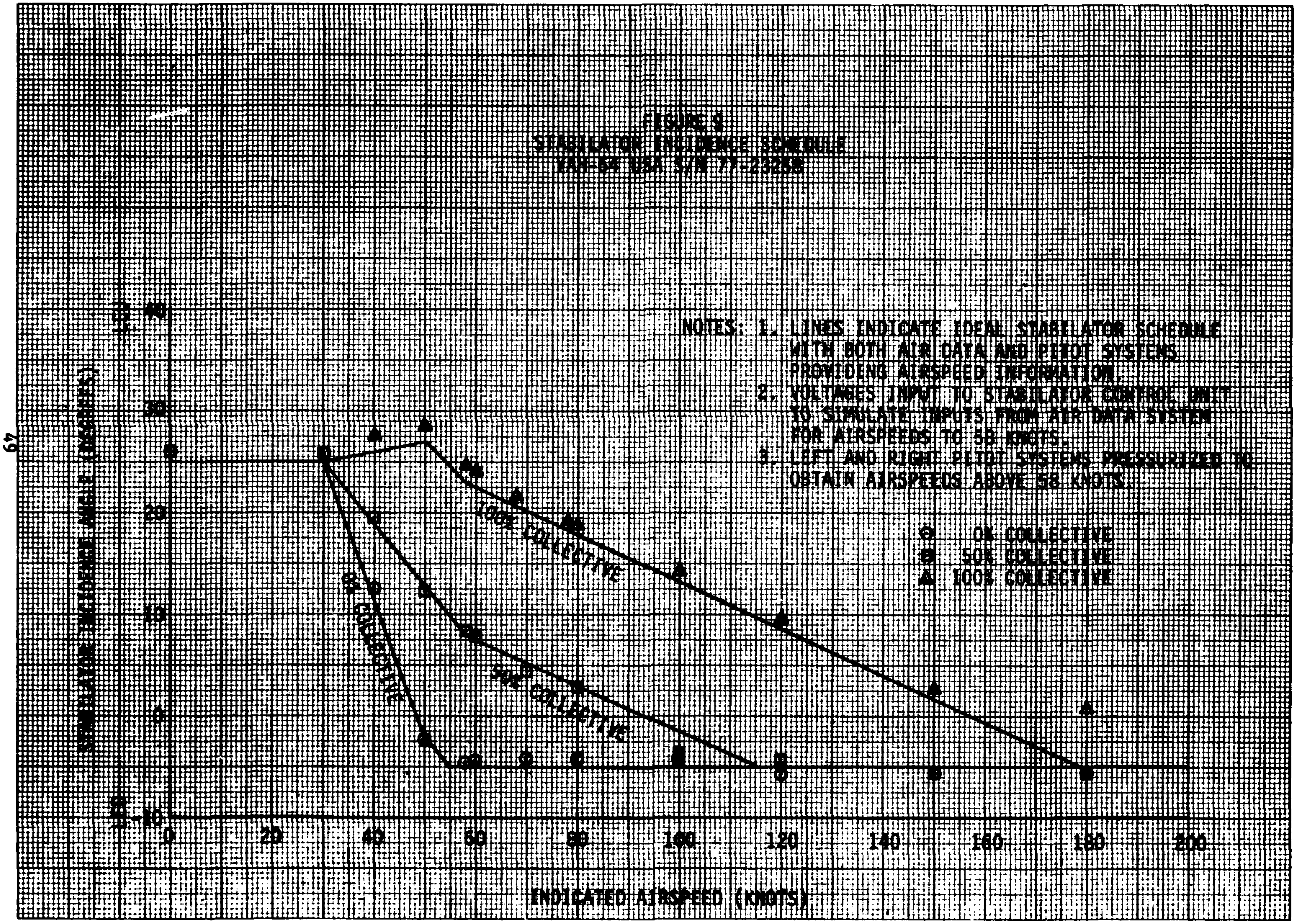


FIGURE 9
STABILATOR INCIDENCE SCHEDULE
YAH-64 USA S/N 77-23258

Table 3. Angle Measurements
Pilots Collective and Cyclic Controls

Blade Azimuth Position (deg)	Item	Rig Pins			Stick position			Measured Clinometer Angle (deg)	Leading Edge Up or Down
		Collective	Longitudinal Cyclic	Lateral Cyclic	Collective	Longitudinal	Lateral		
$\psi = 90$	1	In	In	In	Rig	Rig	Rig	0.2	Down
	2	In	Out	In	Rig	Fwd	Rig	21.0	Down
	3	In	Out	In	Rig	Aft	Rig	11.4	Up
	4	In	In	In	Rig	Rig	Rig	0.4	Down
	5	Out	In	In	Up	Rig	Rig	9.5	Up
	6	Out	In	In	Down	Rig	Rig	9.7	Down
	7	In	In	In	Rig	Rig	Rig	0.4	Down
$\psi = 270$	8	In	In	In	Rig	Rig	Rig	0.4	Down
	9	In	Out	In	Rig	Fwd	Rig	20.8	Up
	10	In	Out	In	Rig	Aft	Rig	11.0	Down
	11	In	In	In	Rig	Rig	Rig	0.9	Up
$\psi = 0$	12	In	In	In	Rig	Rig	Rig	0.5	Down
	13	In	In	Out	Rig	Rig	Left	11.5	Up
	14	In	In	Out	Rig	Rig	Right	7.4	Down
	15	In	In	In	Rig	Rig	Rig	0.3	Down
$\psi = 180$	16	In	In	In	Rig	Rig	Rig	0.0	Down
	17	In	In	Out	Rig	Rig	Left	11.2	Up
	18	In	In	Out	Rig	Rig	Right	7.8	Down
	19	In	In	In	Rig	Rig	Rig	0.1	

Table 4. Computation of Blade Angle Travel Pilots Collective and Cyclic Controls

Computation	Travel (deg)	Tolerance (deg)
LONGITUDINAL CYCLIC		
1. Forward = 1/2 (Item *9 - Item 2)= (If Item 2 is leading edge down add item 2)	20.4	20° (minimum)
2. Aft = 1/2 (Item 3 - Item 10) = (If Item 10 is leading edge down add Item 10)	11.1	10° (minimum)
LATERAL CYCLIC		
3. Left = 1/2 (Item 13 - Item 17) = (If Item 17 is leading edge down add Item 17)	11.4	10.5° (minimum)
4. Right = 1/2 (Item 18 - Item 14) = (If Item 14 is leading edge down add Item 14)	7.8	7.0° (minimum)
COLLECTIVE		
5. Full pitch travel = (Item 5-Item 6)= (If Item 6 is leading edge down add Item 6)	20.0	18.0° (minimum)
6. Measured @ pitch housing (Bolt pad machined surface 2.4 in. inboard of lead-lag hinge)	-8.7	-10° to - 7°

*Item numbers obtained from table 1

Digital Automatic Stabilization Equipment

14. The DASE provides rate damping (SAS), command augmentation (CAS), hover augmentation (HAS), attitude hold, and turn coordination. The DASE is controlled by the digital automatic stabilization equipment computer (DASEC). The DASEC receives information from several sources. The heading and attitude reference system (HARS) provides the DASEC with aircraft angular velocities (3 axes), aircraft attitudes (pitch, roll, and heading), and inertial horizontal and lateral velocities (measured by the Doppler radar). The ADS provides longitudinal airspeed and side-slip angle. The LVDT's provide longitudinal, lateral, collective and directional control position information. The DASEC processes this information and commands control inputs through the electro-hydraulic servo valves on the longitudinal, lateral, and directional servoactuators. The DASE authority is limited in the lateral and directional axis to ± 10 percent of full control authority while the longitudinal axis is limited to 20 percent forward and 10 percent aft. Changes to the DASE since A&FC, Part 1 are presented in table 5. Figure 10 presents a block diagram of the DASE.

Table 5. DASE Modifications Since YAH-64 A&FC Part 1 (S/N 77-23258)

1. Reduction of Yaw SAS actuator travel in response to side slip signal, to 50% of SAS actuator authority.

2. Reduction of side-slip gain by 60% and increase in yaw rate gain of 15%.

3. Reduction of side-slip gain from 1 to 0 when collective is lowered from 50 to 0%.

4. Reduction of HAS gains and inclusion of a 3 second easy ON/OFF for HAS mode.

5. Reduction of roll CAS washout from 20 to 10 sec.

6. Reduction of SAS actuator recentering rate to 0.044 inches of SAS actuator per second in the attitude hold mode.

7. Reprogramming of pitch CAS with airspeed to reduce gain by 50 percent at 160 KTAS.

8. Inclusion of a DASEC low voltage monitor to disengage DASE if power supply low voltage is detected.

9. Increased longitudinal SAS authority to 20% forward and 10% aft.

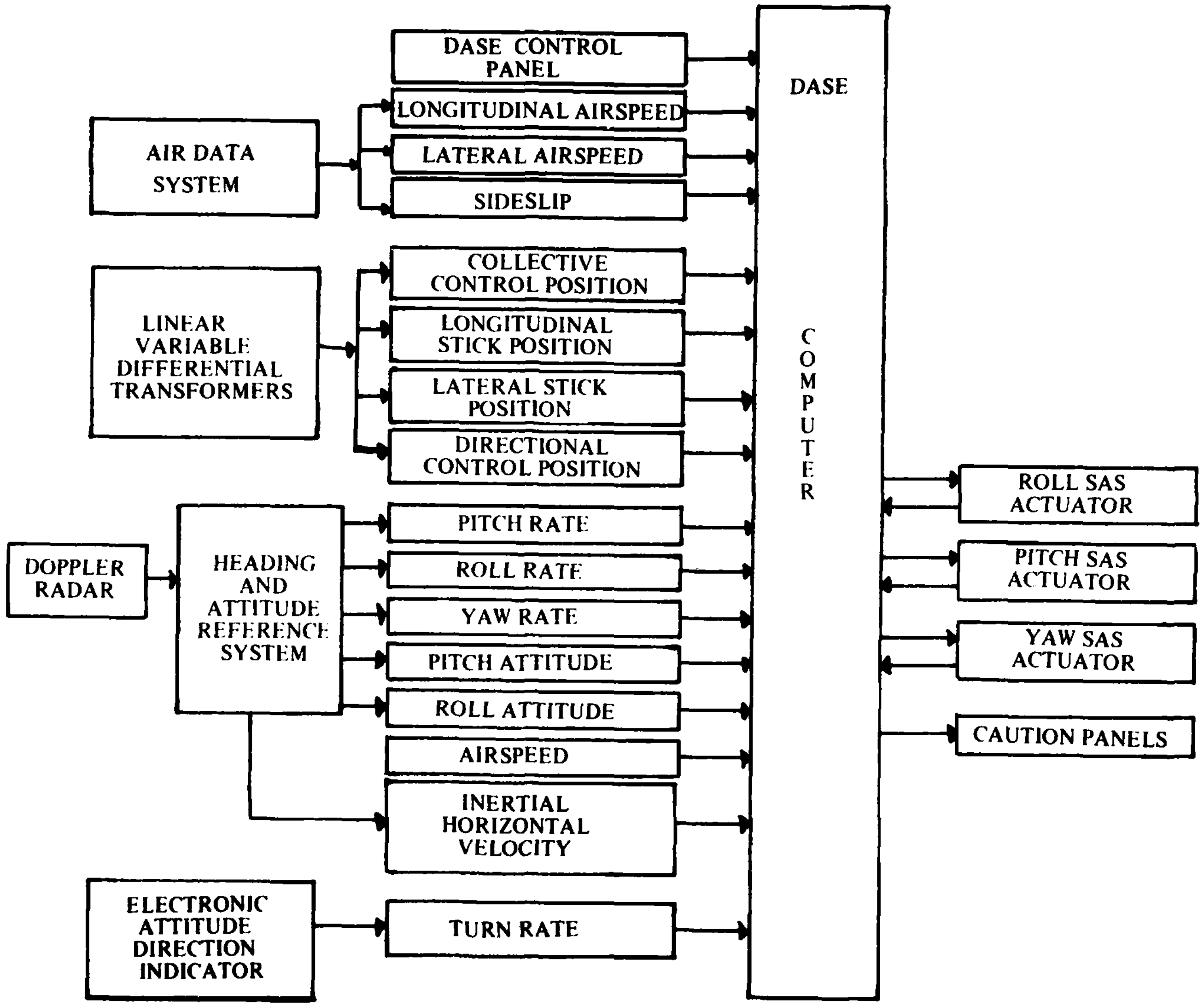


Figure 10. Digital Automatic Stabilization Equipment Block Diagram

15. The SAS function of the DASE system provides rate damping in pitch, roll, and yaw axes. Each axis is separately engageable through a magnetically held toggle switch on the DASE panel shown in figure 11. The CAS is used to augment the pilot control inputs and is an automatic function of the DASE whenever pitch and roll SAS are selected and yaw SAS is selected below 60 KTAS.

16. A limited authority HAS mode is provided through pitch and roll DASE channels using rates, attitudes and Doppler corrected inertial velocities from the HARS. HAS is used to reduce pilot workload in gusty conditions by assisting the pilot in maintaining a desired hover position. HAS is engageable below 15 knots ground speed and 50 KTAS whenever the pitch or roll DASE channels are engaged. Additionally, a heading hold mode is provided through the yaw DASE channel using aircraft heading information from the HARS. This function is engageable whenever the yaw DASE channel is engaged and the HAS switch is selected.

17. A limited authority attitude hold mode is provided through pitch and roll SAS. Attitude Hold is engageable above 60 KTAS whenever pitch and roll SAS are engaged. Attitude Hold will automatically disengage whenever the airspeed is decreased to 50 KTAS.

18. A limited authority turn coordination function is provided through yaw SAS using sideslip information from the ADS. This function is automatically provided above 60 KTAS whenever yaw SAS is engaged.

HYDRAULIC SYSTEM

General

19. The hydraulic system consists of four hydraulic servoactuators powered simultaneously by two independent 3000-psi hydraulic systems. Each servoactuator simultaneously receives pressure from the primary and utility systems to drive the dual-tandem actuators. This design allows the remaining system to automatically continue powering the servos in the event of a single hydraulic system failure. The two systems (primary and utility) are driven by the accessory gearbox utilizing variable displacement pumps, independent reservoirs and accumulators. The APU drives all accessories, including the hydraulic pumps, when the aircraft is on the ground and the rotors are not turning. The accessory gearbox is driven by the main transmission during flight and provides for normal operation of both hydraulic systems during

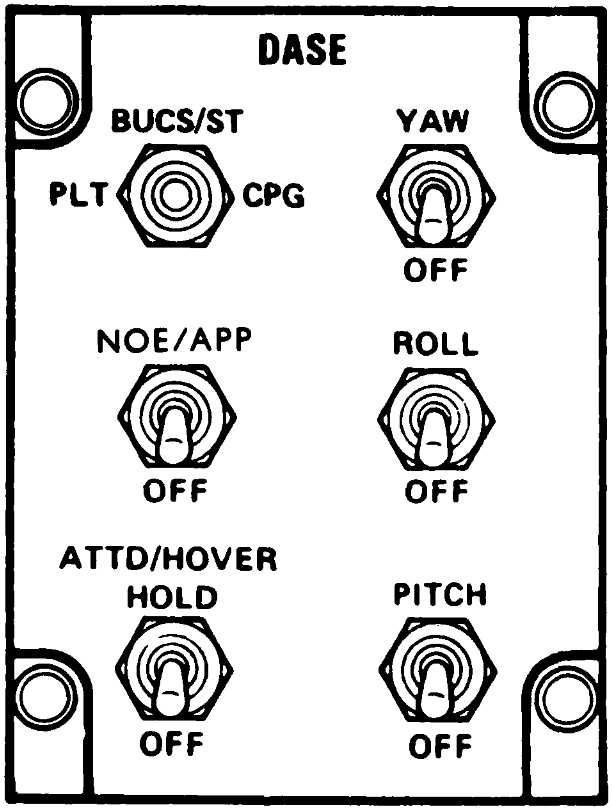


Figure 11. DASE Control Panel

autorotation. An emergency hydraulic system is provided to allow emergency operation of the flight controls in the event of a dual system failure.

Primary Hydraulic System

20. The primary hydraulic system (fig. 12) consists of a one-pint capacity reservoir, which is pressurized to 30 psi using air from the shaft-driven compressor; an accumulator, which has a nitrogen precharge of 1600 psi, designed to reduce surges in the hydraulic system; and a primary manifold that directs the fluid to the lower side of the four serovactuators. The primary system also provides the hydraulic pressure for operation of the DASE functions.

Utility Hydraulic System

21. The utility hydraulic system (fig. 13) consists of an air pressurized 1.3 gallon reservoir and a 3000-psi accumulator which drives the APU starting motor. The utility manifold directs fluid to the upper side of the servoactuators, the stores pylon system, tail wheel lock mechanism, area weapon turret drive, and rotor brake. Other manifold functions include an auxiliary isolation check valve which isolates the area weapon turret drive and external stores actuators when either a low pressure or low fluid condition exists; a low pressure sensor isolates the accumulator as an emergency hydraulic source for the servoactuators in the event of a dual hydraulic system failure. The accumulator assembly stores enough fluid for emergency operation of the flight controls through four full strokes of the collective stick and one 180 degrees heading change. The emergency system may be activated by either the pilot or CPG emergency switch. An electrically activated emergency shutoff valve is designed to isolate the utility side of the directional servoactuator and the tail wheel lock mechanism when a low fluid condition exists. The electrical connections for this valve were not installed on S/N 77-23258 during this test.

Servoactuators

22. Individual hydraulic servoactuators are provided for longitudinal, lateral, collective, and directional controls. Each servoactuator (fig. 14) consists of a ballistically tolerant housing, a single actuator rod and dual frangible pistons, a BUCS plunger, and various parts for routing of both primary and utility hydraulic fluid. The system is designed to accomodate all flight loads with a failure of either system. DASE and BUCS functions would be lost with failure of the primary

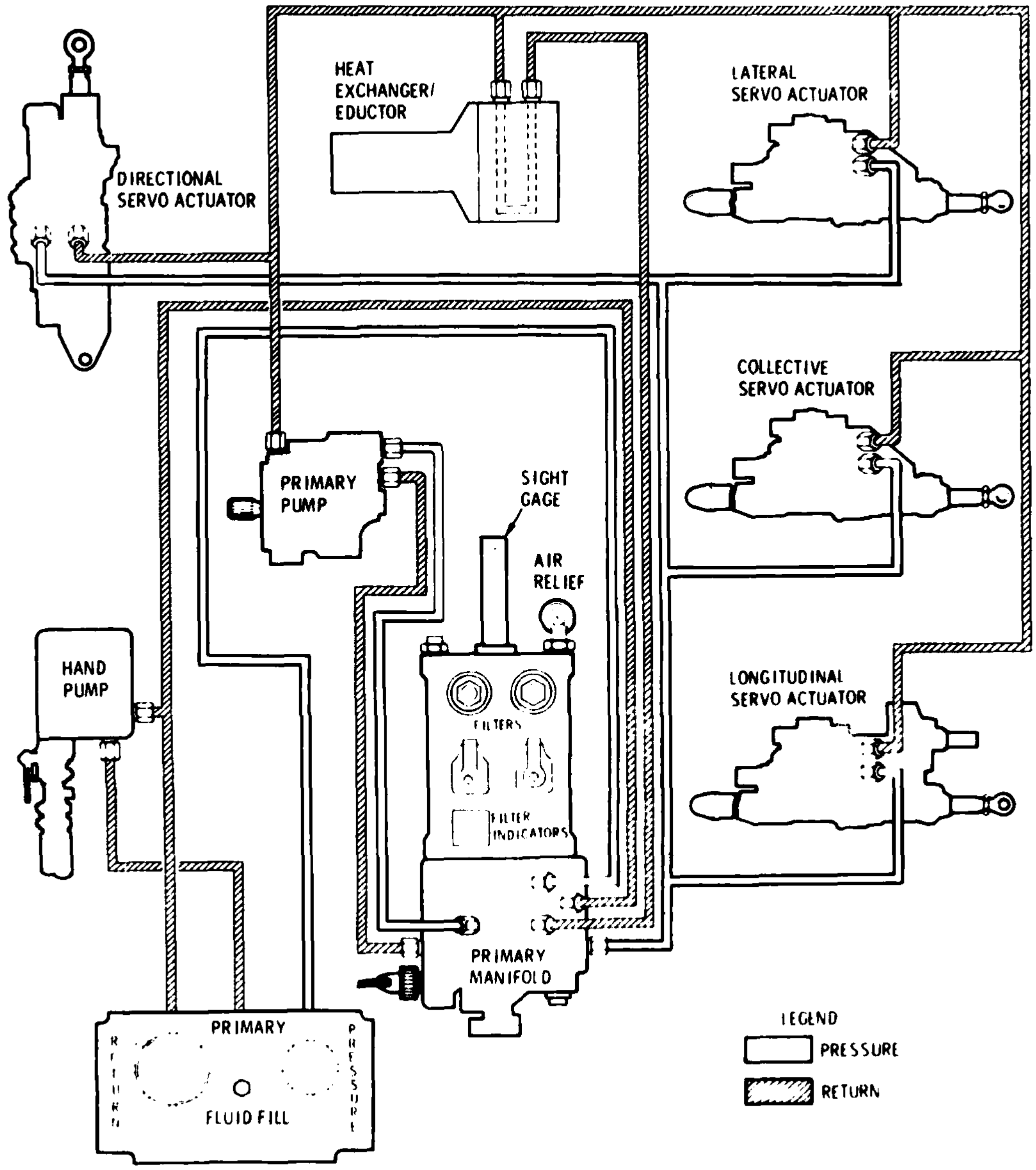


Figure 12. Primary Hydraulic System

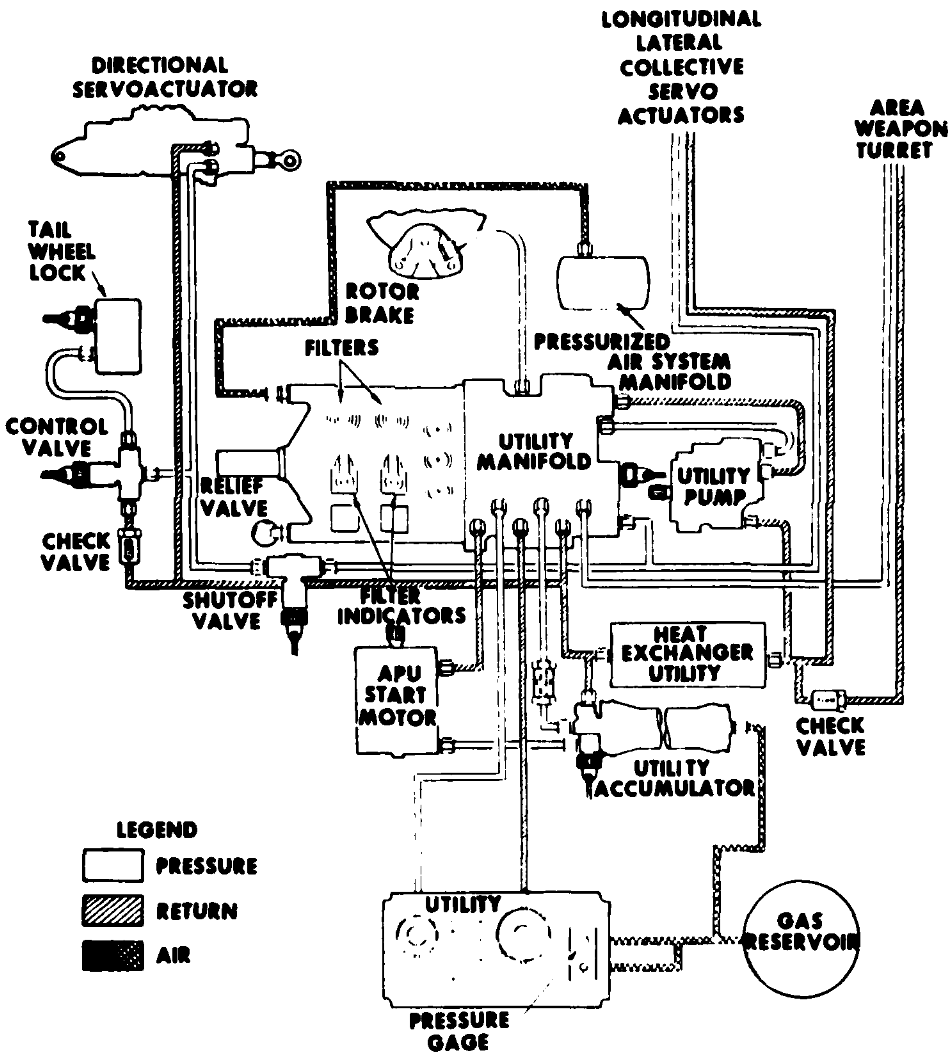


Figure 13. Utility Hydraulic System

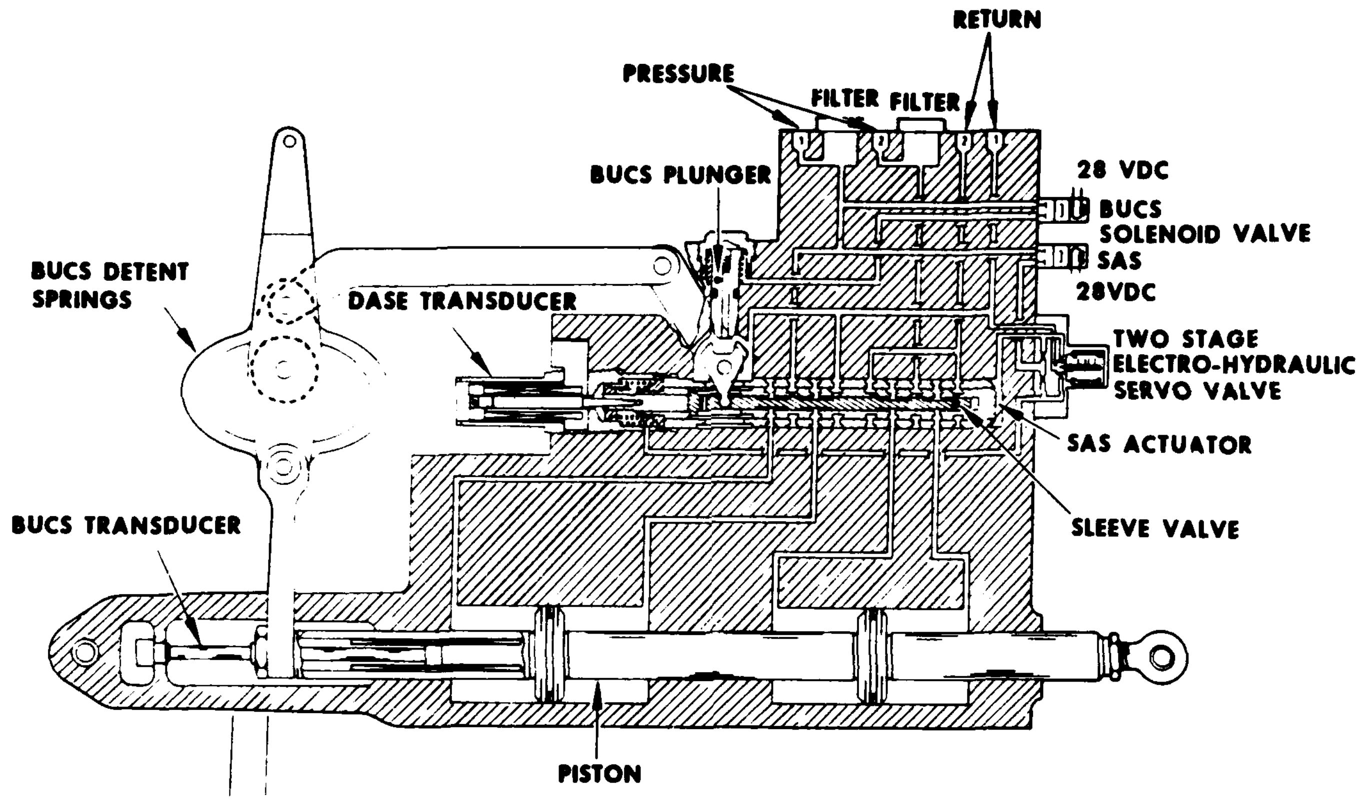


Figure 14. Flight Control Servoactuator

system. The BUCS plunger assemblies were installed during this test, however, electrical connections were omitted.

POWER PLANT

23. The power plant installed in the YAH-64 helicopter for this test was the General Electric YT700-GE-700R front drive turboshaft engine, rated at 1563 shp (sea level, standard day, uninstalled). The engines are mounted in nacelles on either side of the main transmission. The basic engine consists of four modules: A cold section, a hot section, a power turbine, and an accessory section. Design features of each engine include an axial-centrifugal flow compressor, a through-flow combustor, a two-stage air-cooled high-pressure gas generator turbine, a two-stage uncooled power turbine and self-contained lubrication and electrical systems. In order to reduce sand and dust erosion, and foreign object damage, an integral particle separator operates when the engine is running. The YT700-GE-700R engine also incorporates a history recorder which records total engine events. Engines S/N 207-239 and 207-258R were installed in the left and right positions, respectively. Both engines were modified since A&FC, Part 1 by replacement of the G07 electrical control unit (ECU) with G04HB units. Pertinent engine data are shown:

Model	YT700-GE-700R
Type	Turboshaft
Rated power (intermediate)	1563 shp sea level, standard day, uninstalled
Output speed (at 100 percent N_R)	20,952 RPM
Compressor	5 axial stages, 1 centrifugal stage
Variable geometry	Inlet guide vanes, stages 1 and 2 Stator vanes
Combustion chamber	Single annular chamber with axial flow
Gas generator turbine stages	2
Power turbine stages	2

Direction of rotation (aft looking forward)	Clockwise
Weight (dry)	415 lb
Length	46.5 in
Maximum diameter	25 in
Fuel	MIL-T-5624 (JP-4 or JP-5)
Lubricating oil	MIL-L-7808 or MIL-L-23699
Electrical power requirements for history recorded and N_p overspeed protection	40W, 115 VAC, 400 Hz
Electrical power requirements for anti-ice valve, filter bypass indication, oil filter bypass indication, and magnetic chip detector	1 amp, 28 VDC

INFRARED (IR) SUPPRESSION SYSTEM

24. The IR suppression system consists of finned exhaust pipes attached to the engine outlet and bent outboard to mask hot engine parts. The finned pipes radiate heat which is cooled by rotor downwash in hover and turbulent air flow in forward flight. The engine exhaust plume is cooled by mixing it with engine cooling air and bay cooling air (fig. 15). The exhaust acts as an eductor, creating air flow over the combustion section of the engine providing engine cooling. Fixed louvers on the top and bottom of the aft cowl and a door on the bottom forward cowling provide convective cooling to the engine during shutdown. The movable bottom door is closed by engine bleed air during engine operation.

FUEL SYSTEM

25. The YAH-64 fuel system has two fuel cells located fore and aft of the ammunition bay. The system includes a fuel boost pump in the aft cell for starting and for high-altitude operation, a fuel transfer pump for transferring fuel between cells, a fuel crossfeed/shutoff valve, and provisions for pressure and gravity

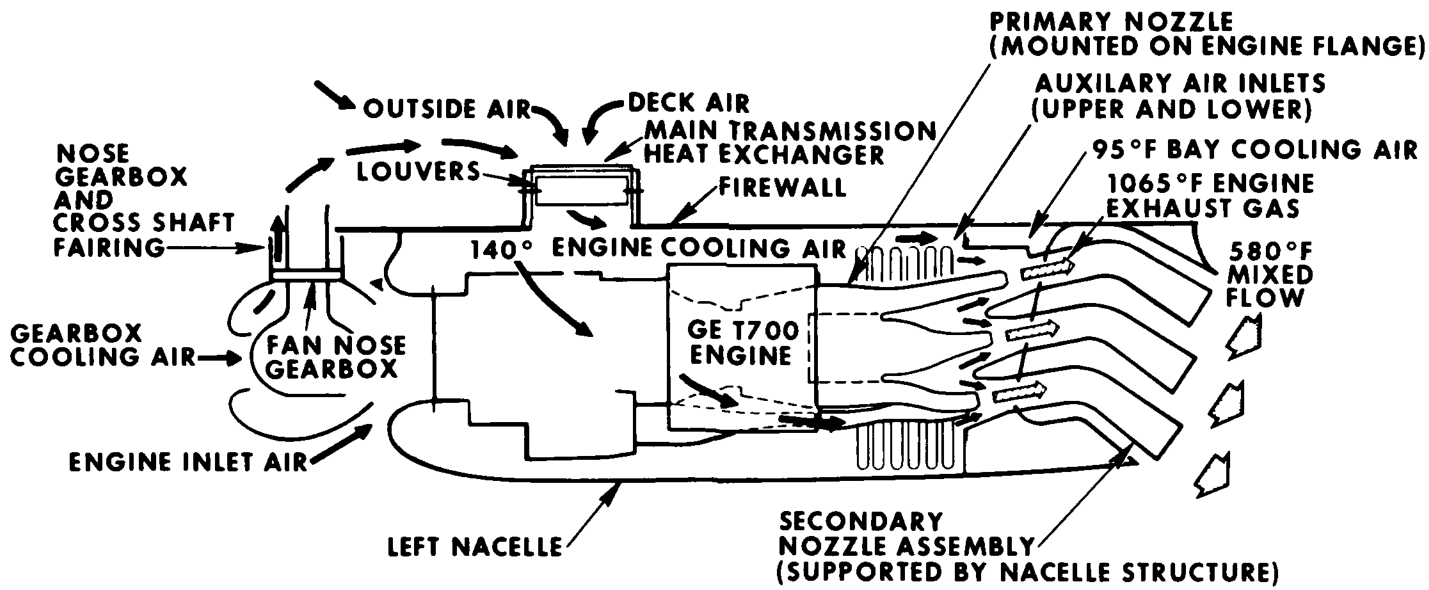


Figure 15. Infrared Suppression System Engine Cooling

fueling and defueling. Additionally, provisions exist for external, wing-mounted fuel tanks. Figure 16 is a schematic of the fuel system. Figure 17 shows the locations and capacities of the two internal fuel cells.

26. By using the tank select switch on the fuel control panel (fig. 18), the pilot or CPG can select either or both tanks from which the engines will draw fuel. With the tank select switch in the NRML position, the left (No. 1) engine will draw fuel from the forward fuel cell and the right (No. 2) engine will draw from the aft cell. When FROM FWD is selected on the tank select switch, the two fuel crossfeed/ shutoff valves are positioned so that both engines draw fuel from the forward tank. The FROM AFT position allows the engines to draw fuel from the aft tank only. The tank select switch is disabled whenever the boost pump is on. When the boost pump is on, the fuel crossfeed/shutoff valves are positioned to allow only fuel from the aft cell to feed both engines. The airdriven boost pump operates automatically during engine start and may be activated by the switch on the pilot or CPG fuel control panel.

27. The pilot and CPG also have the capability to transfer fuel between tanks using the transfer switch on the fuel control panels. Moving the fuel transfer switch out of the OFF position closes the refuel valve and activates the air-driven pump which transfers fuel in the selected direction.

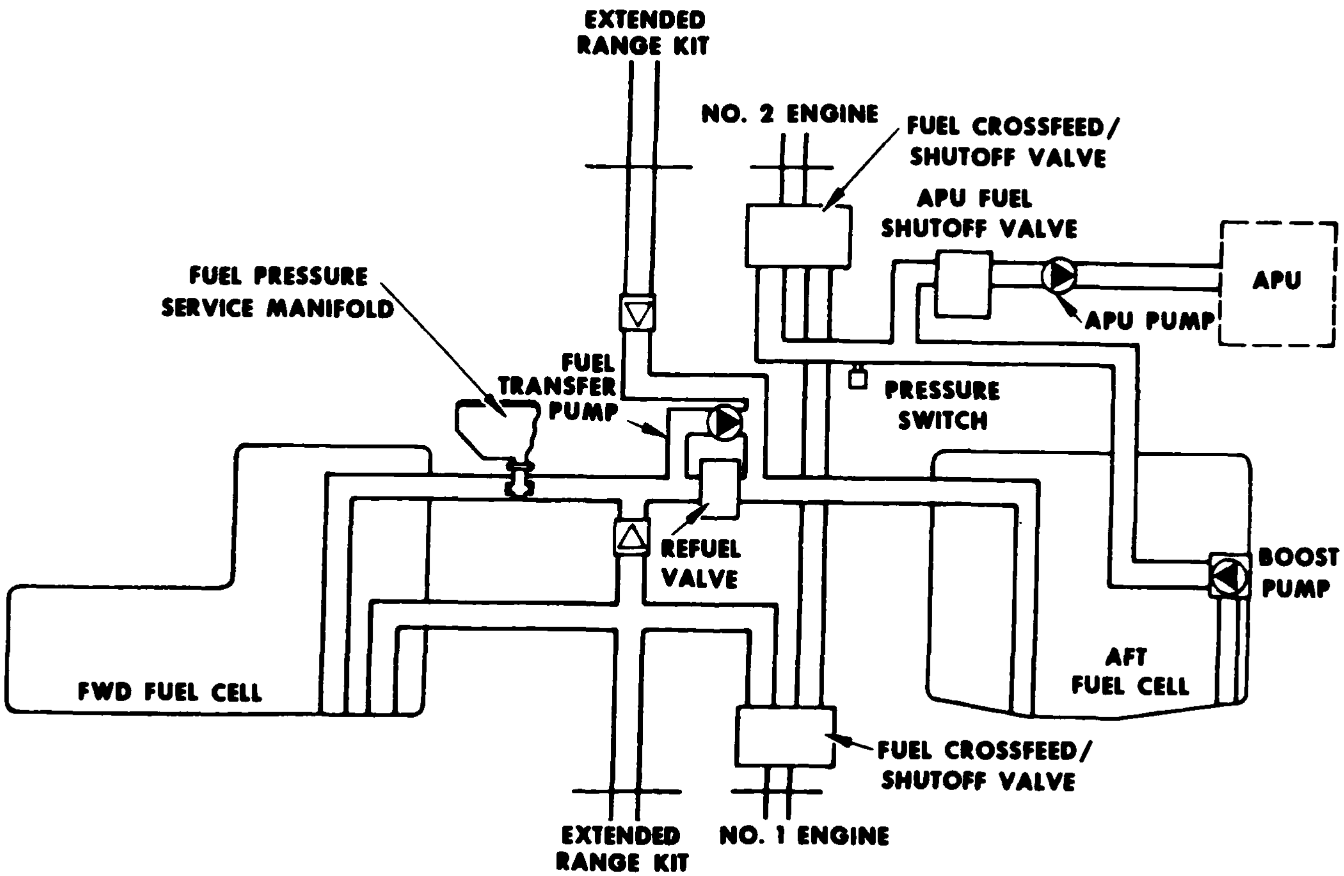


Figure 16. Fuel System Major Components

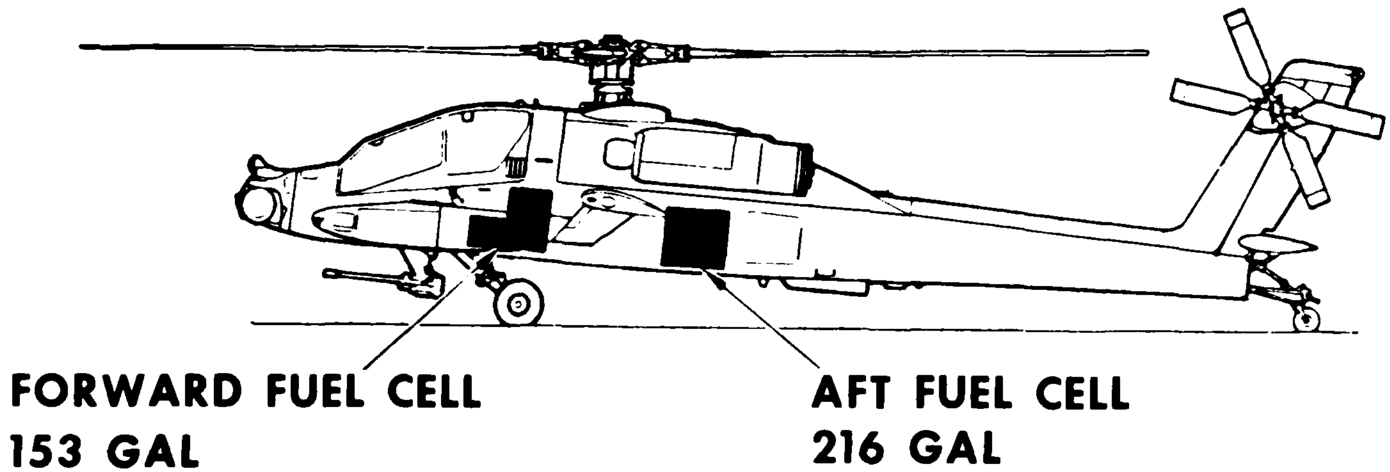


Figure 17. Fuel Cell Location

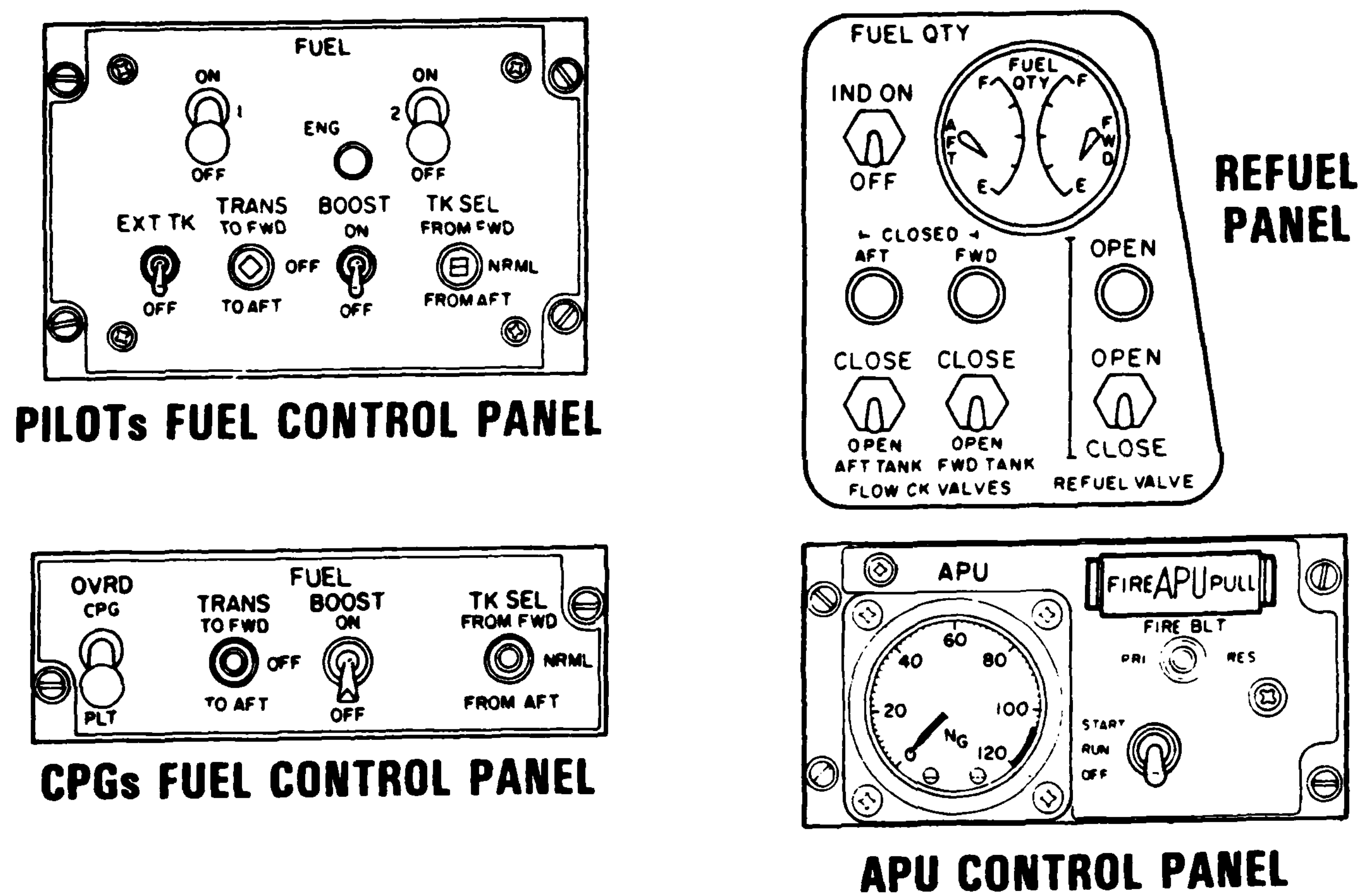


Figure 18. Fuel System Controls

APPENDIX C. INSTRUMENTATION

The airborne data acquisition system was installed, calibrated, and maintained by Hughes Helicopters Incorporated. The system used pulse code modulation (PCM) encoding, and magnetic tape was used to record parameters on board the aircraft. A boom was mounted on the left side of the aircraft, extending 52 inches forward of the nose. A pitot-static tube, an angle-of-attack sensor, and an angle-of-sideslip sensor were mounted on the boom. Calibration of the boom airspeed system is presented in figure 1 through 3. Instrumentation and related special equipment used for this test follows:

<u>Pilot Station (Aft Cockpit)</u>

Pressure altitude (ship)
Airspeed (boom) (sensitive)
Air data system low airspeed (lateral and longitudinal)
Main rotor speed (digital)
Engine torque (both engines)*
Engine turbine gas temperature (both engines)*
Engine power turbine speed (both engines)*
Engine gas producer speed (both engines)*
Angle of sideslip
Event switch
Longitudinal control position
Lateral control position
Directional control position
Collective control position
Stabilator angle*
Normal acceleration (CG)

<u>Copilot/Engineer Station</u>

Airspeed (left pitot)
Altitude (ship)
Main rotor speed
Engine torque (both engines)*
Engine turbine gas temperature (both engines)*
Engine gas producer speed (both engines)*
Total air temperature
Time code display
Event switch
Data system controls
Fuel used (both engines)

*standard ship system

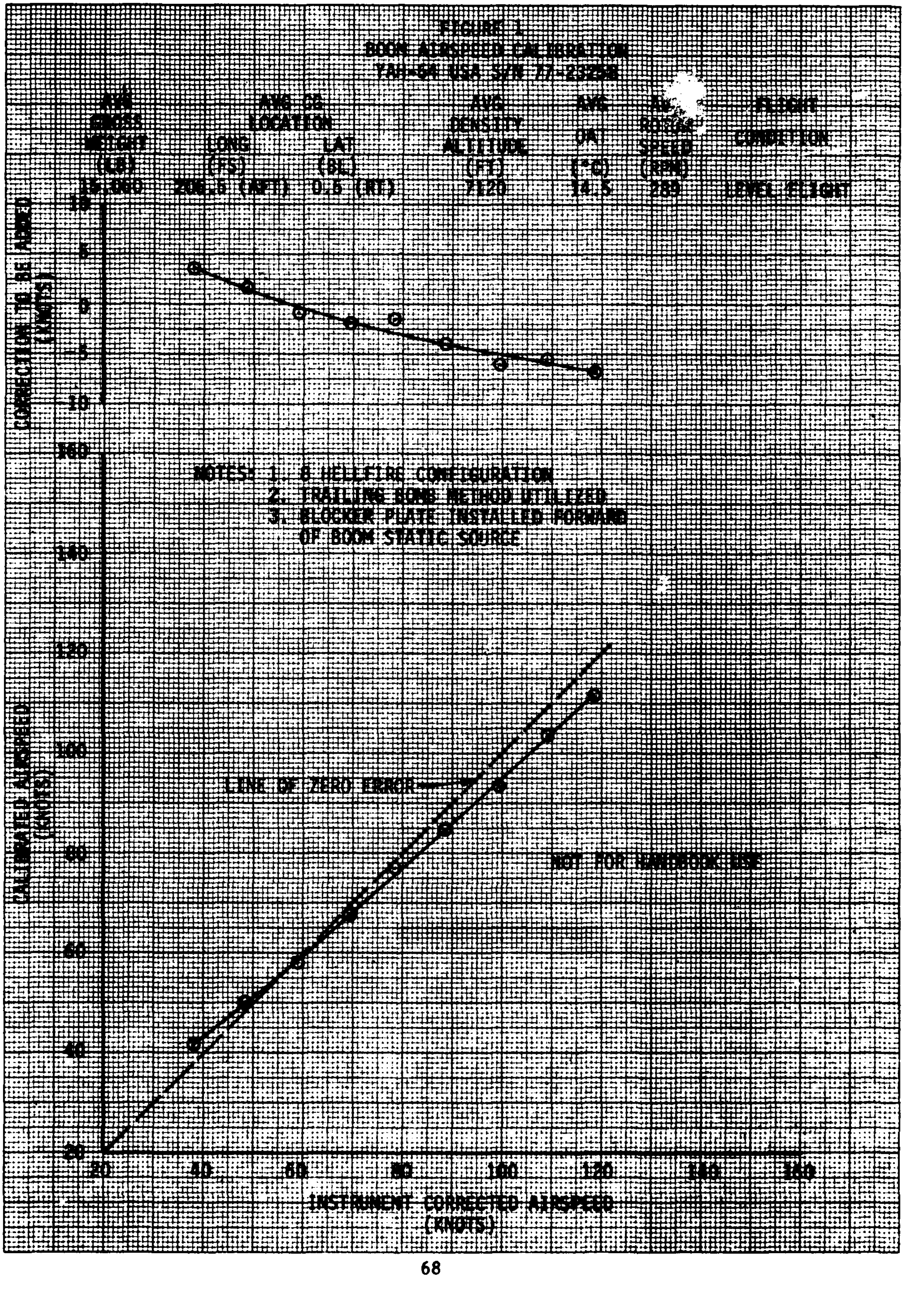


FIGURE 1
BOOM AIRSPEED CALIBRATION
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB) 15,000
AVG CG LOCATION LONG (FS) 206.5 (AFT) LAT (BL) 0.6 (RT)
AVG DENSITY ALTITUDE (FT) 7120
AVG OAT (°C) 14.5
AVG ROTOR SPEED (RPM) 289
FLIGHT CONDITION LEVEL FLIGHT
CORRECTION TO BE ADDED (KNOTS)
NOTES: 1. 8 HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD UTILIZED
3. BLOCKER PLATE INSTALLED FORWARD OF BOOM STATIC SOURCE
LINE OF ZERO ERROR
NOT FOR HANDBOOK USE
CALIBRATED AIRSPEED (KNOTS)
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

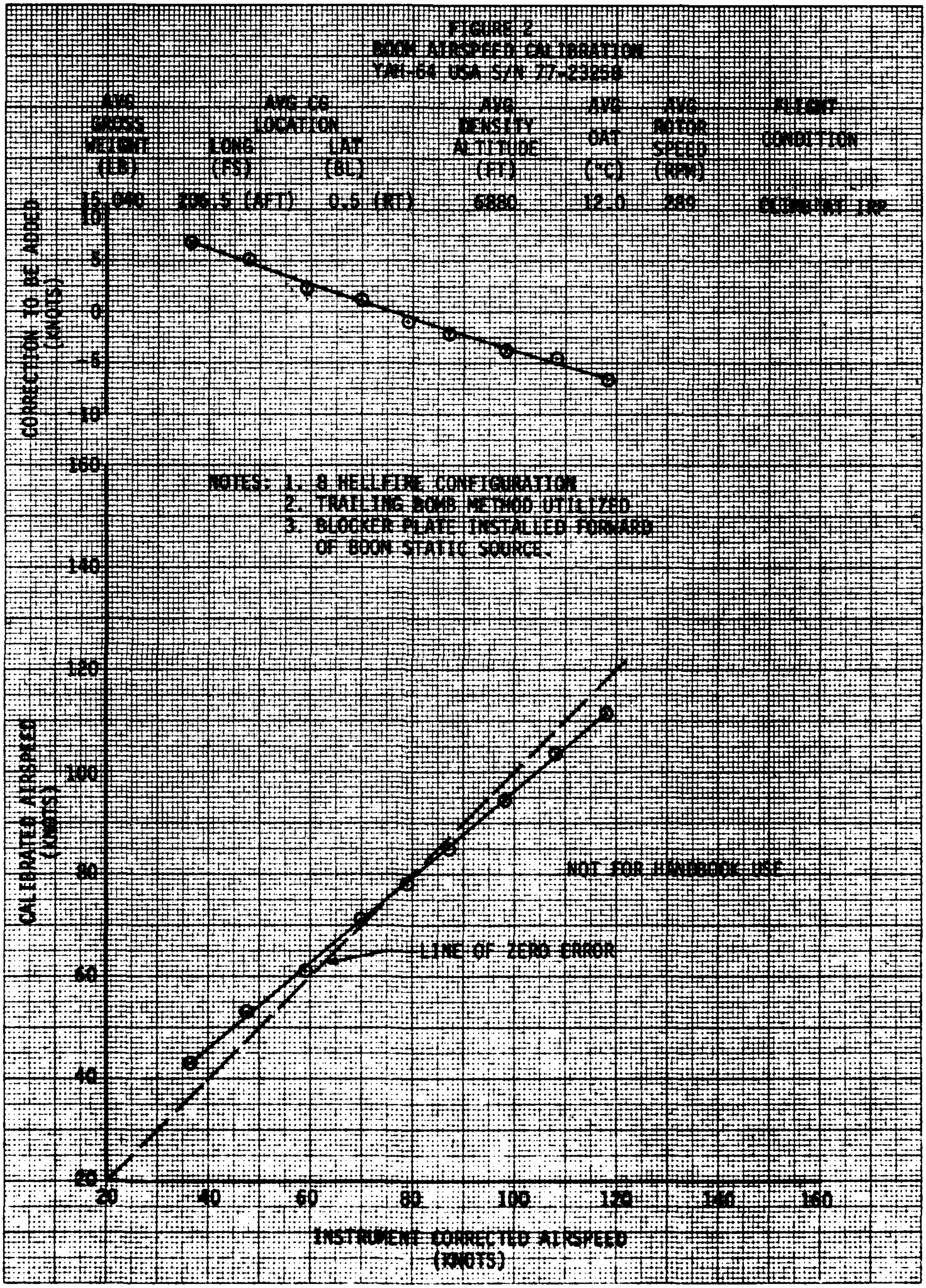


FIGURE 2
BOOM AIRSPEED CALIBRATION
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15,040
206.5 (AFT)
0.5 (RT)
6880
12.0
289
CORRECTION TO BE ADDED (KNOTS)
10
5
0
-5
-10
NOTES: 1. 8 HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD UTILIZED
3. BLOCKER PLATE INSTALLED FORWARD OF BOOM STATIC SOURCE.
CALIBRATED AIRSPEED (KNOTS)
160
140
120
100
80
60
40
20
NOT FOR HANDBOOK USE
LINE OF ZERO ERROR
20
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

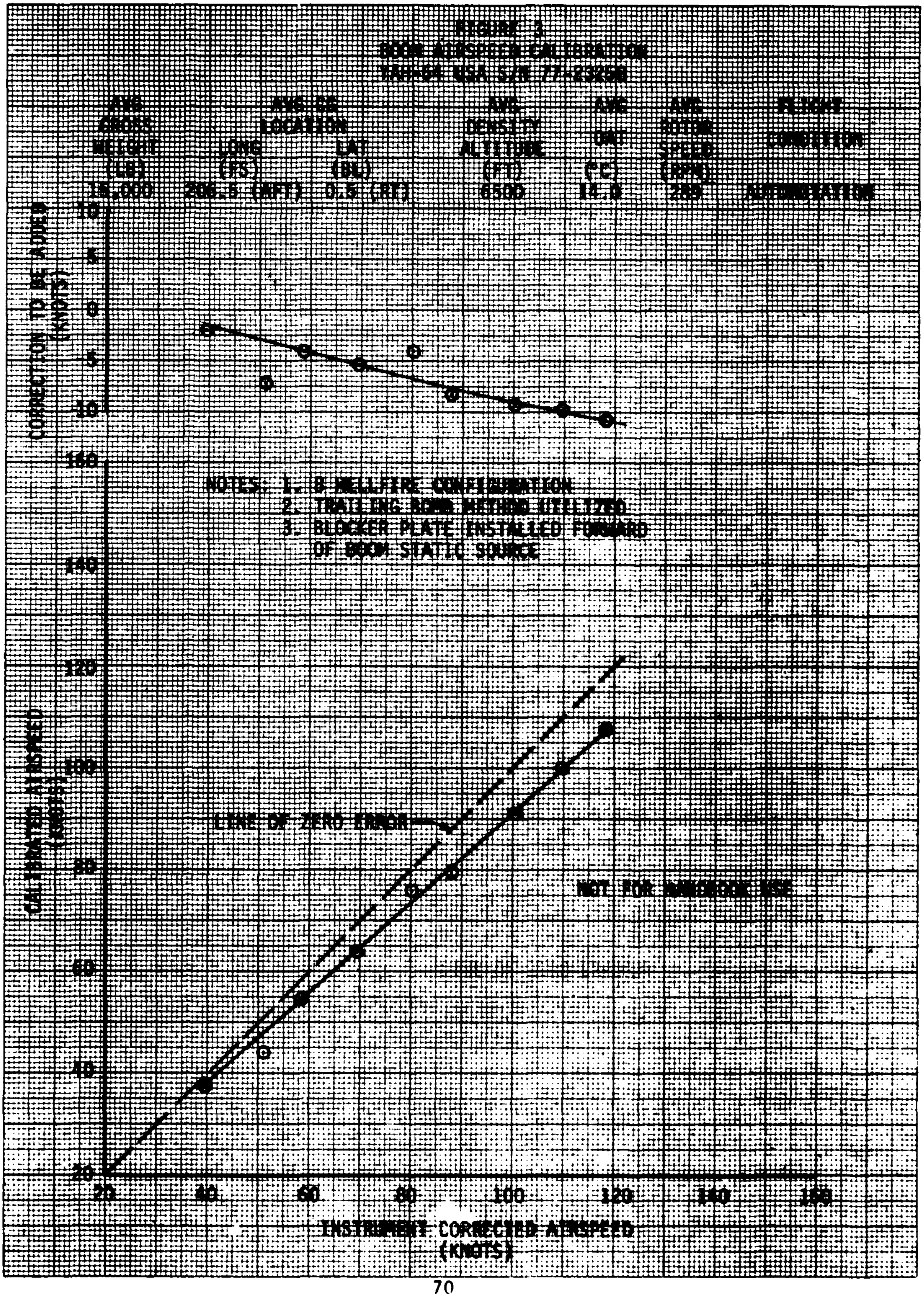


FIGURE 3
BOOM AIRSPEED CALIBRATION
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB) 15,000
AVG CG LOCATION LONG (FS) 206.5 (AFT) LAT (BL) 0.5 (RT)
AVG DENSITY ALTITUDE (FT) 6500
AVG OAT (°C) 14.0
AVG ROTOR SPEED (RPM) 289
FLIGHT CONDITION AUTOROTATION
CORRECTION TO BE ADDED (KNOTS)
NOTES: 1. 8 HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD UTILIZED
3. BLOCKER PLATE INSTALLED FORWARD OF BOOM STATIC SOURCE
LINE OF ZERO ERROR
NOT FOR HANDBOOK USE
CALIBRATED AIRSPEED (KNOTS)
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

PCM Parameters

Time code
Event
Airspeed (boom)
Airspeed (ship, right and left)
Altitude (boom)
Altitude (ship)
Total air temperature
Main rotor speed
Fuel temperature (both engines)
Fuel used (both engines)
Engine fuel flow rate (both engines)
Engine gas producer speed (both engines)
Engine power turbine speed (both engines)
Engine torque (both engines)
Engine turbine gas temperature (both engines)
Angle of attack
Angle of sideslip
Control positions:
- Longitudinal cyclic
- Lateral cyclic
- Directional
- Collective

Aircraft attitudes:
- Pitch
- Roll
- Yaw

Aircraft angular velocities:
- Pitch
- Roll
- Yaw

Stability augmentation system actuator position:
- Longitudinal
- Lateral
- Directional

Control actuator positions:
- Tail rotor
- Collective pitch
- Longitudinal cyclic
- Lateral cyclic

Air data system:
- Longitudinal airspeed
- Lateral airspeed
- Resultant airspeed
- Pressure altitude
- Air Temperature

Radar altitude

Center of gravity normal acceleration
Vibration accelerometers:
    Pilot seat (3 axes)
    Pilot floor (3 axes)
    Copilot seat (3 axes)
    Copilot floor (3 axes)
    Aircraft center of gravity (3 axes)

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. The helicopter performance test data were generalized by use of nondimensional coefficients and were such that the effects of compressibility and blade stall were not separated and defined. The following nondimensional coefficients were used to generalize the hover and vertical climb test results obtained during this flight test program.

a. Coefficient of power (C_P):

$$C_P = \frac{SHP\ (550)}{\rho A(\Omega R)^3} \tag{1}$$

b. Coefficient of thrust (C_T):

$$C_T = \frac{Thrust}{\rho A(\Omega R)^2} \tag{2}$$

c. Generalized coefficient of Power (C_{PGEN})

$$C_{PGEN} = \frac{C_{P_C} - C_{P_H}}{0.707\ C_T^{1.5}} \tag{3}$$

d. Non dimensional vertical velocity ($\overline{V}_V$)

$$\overline{V}_v = (R/C)/\Omega R\sqrt{C_T}/2 \tag{4}$$

Where:

SHP = Engine output shaft horsepower (both engines)
550 = Conversion factor (ft-lb/sec)/SHP
ρ = Air density (slug/ft^3)
A = Main rotor disc area (ft^2) = (180./56)
Ω = Main rotor angular velocity (radian/sec) = 30.26 (at 289 RPM)
R - Main rotor radius (ft) = 24
Thrust = Gross weight (lb) during free flight in which there is no acceleration or velocity component in the vertical direction.
C_{P_C} = Coefficient of power used during vertical climb

C_{P_H} = Coefficient of power required to hover out of ground effect at specified C_T
R/C = Rate of climb (ft/sec)

For a rotor speed of 289 RPM, the following constants were used:

$A = 1809.56\ ft^2$
$\Omega R = 726.34\ ft/sec$
$A(\Omega R)^2 = 954657879\ ft^4/sec^2$
$A(\Omega R)^3 = 6.934025959 \times 10^{11}\ ft^5/sec^3$

SHAFT HORSEPOWER REQUIRED

2. Engine output shaft torque was determined by the use of the engine torquemeter. The torquemeter was calibrated in a test cell by the engine manufacturer prior to Part 1 of the A&FC flight tests (June 1981). During Part 1 engine serial number 207258R malfunctioned, (foreign object damage to compressor), was replaced for those tests and subsequently repaired and used for Part 2 of the A&FC tests. Engine serial number 207239R was used during both Part 1 & 2 of the A&FC. The electrical control units of each engine were replaced since their calibration in the test cell but no data was available to apply the necessary corrections. Hover data was obtained in conjunction with the vertical climb tests to evaluate the engine torquemeter calibrations. These data shown in figure 1 appendix E compared favorably to previous test results (refs 4 and 6, app A). Since the torquemeter data was used to determine a test day power required difference between hover and vertical climb, the torquemeter calibrations were considered satisfactory and were used as originally supplied by the engine manufacturer. The output from the engine torquemeters was recorded on the onboard data recording system. The output shaft horsepower was determined from the engine shaft torque and rotational speed by the following equation:

$$SHP = \frac{2\pi \times N_p \times Q}{33{,}000}$$

Where:

N_p = Engine output shaft rotational speed (RPM)

Q = Calibrated engine output shaft torque (ft-lb)
33,000 = Conversion factor (ft-lb/min/SHP)

VERTICAL CLIMB PERFORMANCE

3. Vertical climb performance data were obtained by stabilizing the helicopter in a 100-foot OGE hover, based on a radar altimeter indication, increasing engine power by a predetermined increment of engine torque, and allowing vertical rate of climb to stabilize. Various power increments up to the engine limit were used. Data were recorded during the stabilized hover and when the helicopter attained an unaccelerating vertical climb. The ship's Low Airspeed System (Air Data System manufactured by Pacer, Inc.) was used to provide cues of longitudinal and lateral translation during the climbs. The ratio of main rotor speed to the square root of the temperature ratio was held constant by averaging the ambient air temperature in the altitude test band and setting the appropriate actual main rotor speed prior to initiating the vertical climb. The ratio of gross weight to density ratio was allowed to vary. Tests were conducted in winds of 3 knots or less.

4. The climb rates were measured after the helicopter was stabilized in unaccelerating vertical climbing flight by differentiating the recorded output of the radar altimeter with respect to time. Vertical climb performance was determined nondimensionally in terms of vertical velocity ratio and generalized power coefficient ($C_{P_{GEN}}$) shown in equations c and d. $C_{P_{GEN}}$ was calculated using the curve through the hover data shown in figure 1, appendix E, as the basis for the OGE hover power required. The curve was obtained by a comparison with previous test results (refs 4 and 6, app A) momentum theory considerations of the nondimensional vertical climb data and the test data obtained. This method was used in an attempt to minimize systematic instrumentation or engine torquemeter calibration errors.

5. Previous test results (A&FC Part 1) showed that the ADS was not a reliable indication of rearward airspeed. A low airspeed calibration was performed using a calibrated pace vehicle and correcting for surface winds prior to conducting the vertical climbs. Data were obtained for longitudinal airspeed only and indicate a position error band of approximately ± 5 knots from 35 knots rearward to 35 knots forward speeds (fig. 1). This amount of scatter is not satisfactory for test purposes, when used as the only speed reference, and the ADS in its present configuration should not be used for low speed test data. However, the ADS indications tempered by the atmospheric and light wind conditions (during many of the climbs calm surface winds were recorded) were used as a horizontal speed reference and were considered satisfactory for these vertical climb tests.

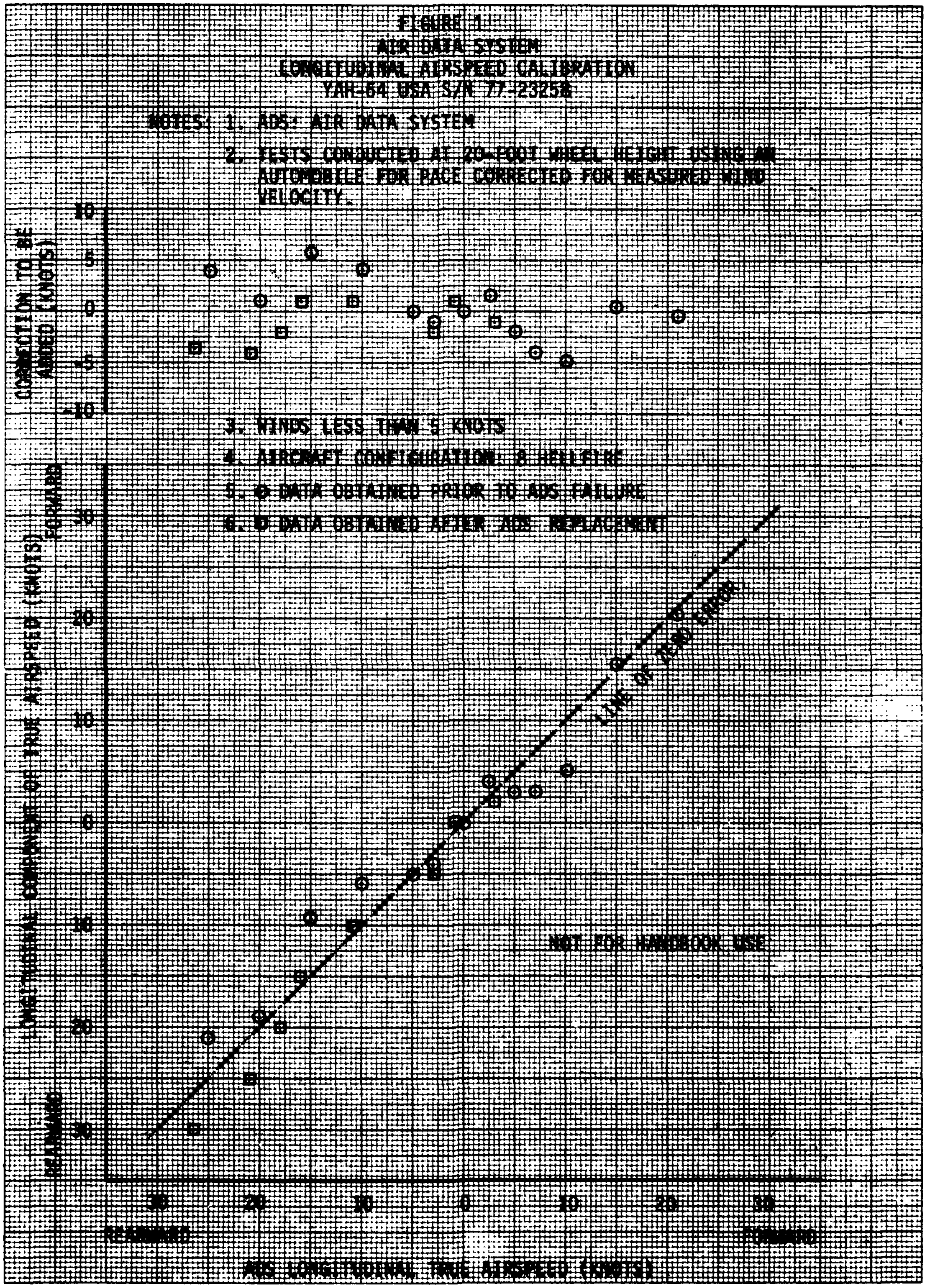


FIGURE 1
AIR DATA SYSTEM
LONGITUDINAL AIRSPEED CALIBRATION
YAH-64 USA S/N 77-23258
NOTES: 1. ADS: AIR DATA SYSTEM
2. TESTS CONDUCTED AT 20-FOOT WHEEL HEIGHT USING AN AUTOMOBILE FOR PACE CORRECTED FOR MEASURED WIND VELOCITY.
3. WINDS LESS THAN 5 KNOTS
4. AIRCRAFT CONFIGURATION: 8 HELLFIRE
5. ⊙ DATA OBTAINED PRIOR TO ADS FAILURE
6. ⊡ DATA OBTAINED AFTER ADS REPLACEMENT
CORRECTION TO BE ADDED (KNOTS)
LONGITUDINAL COMPONENT OF TRUE AIRSPEED (KNOTS)
FORWARD
REARWARD
LINE OF ZERO ERROR
NOT FOR HANDBOOK USE
ADS LONGITUDINAL TRUE AIRSPEED (KNOTS)
REARWARD
FORWARD
10
5
0
-5
-10
30
20
10
0
10
20
30
30
20
10
0
10
20
30

HANDLING QUALITIES

6. Stability and control data were collected and evaluated using standard test methods as described in reference 14, appendix A. The Handling Qualities Rating Scale (HQRS) presented in figure 2 was used to augment pilot comments relative to handling qualities and work load.

VIBRATIONS

7. The vibration data were reduced by means of a fast Fourier transform from the analog flight tape averaged over a period of 20 seconds. Vibration levels, representing peak amplitudes, were extracted from this analysis at selected harmonics of the main rotor frequency. The Vibration Rating Scale (VRS), presented in figure 3, was used to augment crew comments on aircraft vibration levels.

AIRSPEED CALIBRATION

8. The boom pitot-static system, the air data system and both ship's systems were calibrated in flight regimes including climbs at IRP, climbs at 1000 FPM rate of climb, level flight, and autorotation. The trailing bomb method was used to determine the airspeed position error. The trailing bomb installation is shown in photos 1 and 2. The air data system was also calibrated in low speed forward and rearward flight using the pace vehicle method (para 5). Calibrated airspeed (V_{cal}) was obtained by correcting indicated airspeeds (V_{in}) using instrument (ΔV_{ic}) and position (ΔV_{pc}) error corrections.

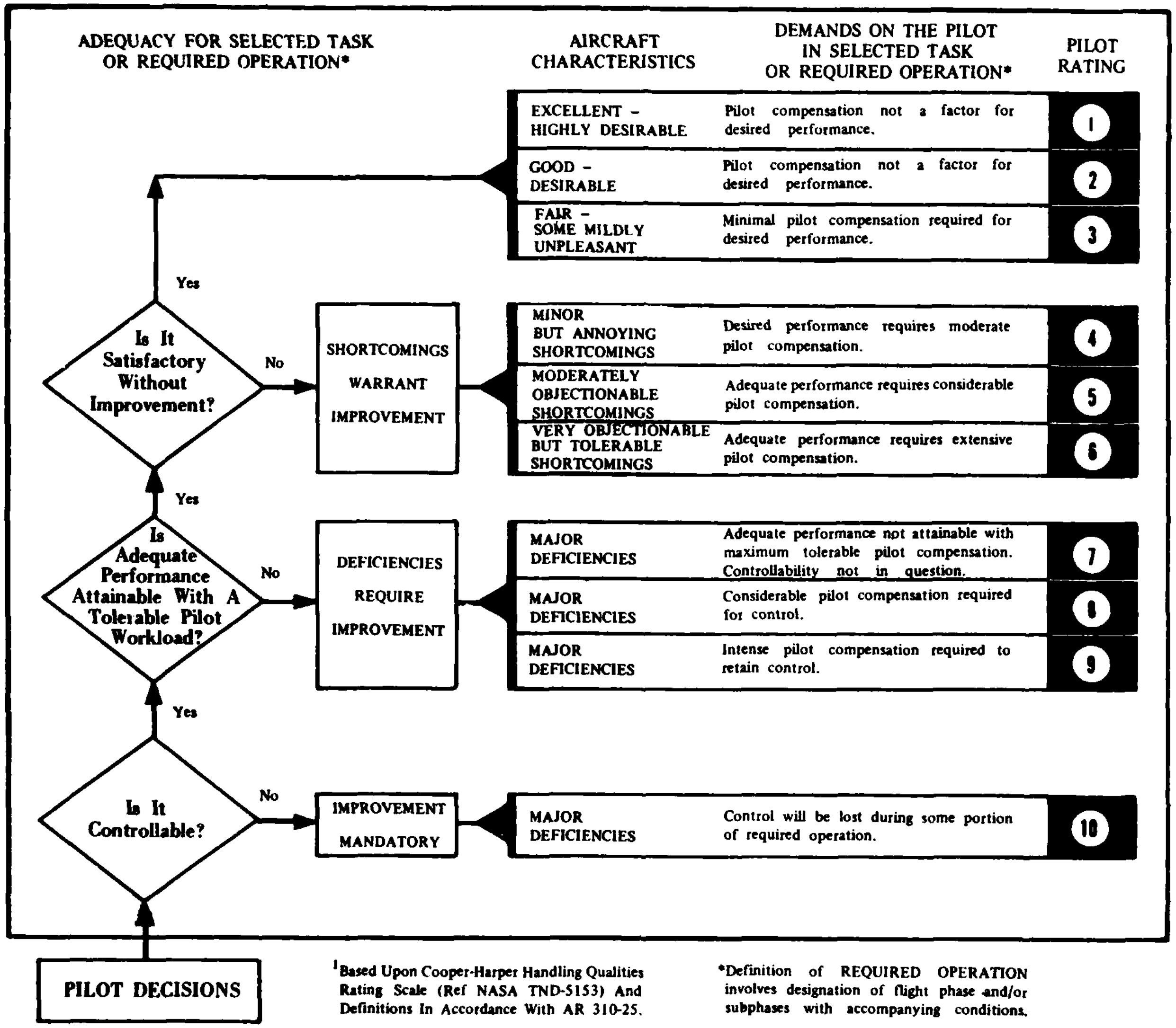
$$V_{cal} = V_{in} + \Delta V_{ic} + \Delta V_{pc}$$

WEIGHT AND BALANCE

9. Prior to testing, the helicopter gross weight and CG were determined using calibrated scales. The aircraft was weighed with trapped fuel, in the clean configuration with test instrumentation on board. The aircraft weight was 11,939 pounds with a longitudinal CG location at FS 213.7 and lateral CG location at BL 0.6.

DEFINITIONS

10. The following definitions of deficiencies and shortcomings were used during the evaluation.



[1] Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Figure 2. Handling Qualities Rating Scale

DEGREE OF VIBRATION	DESCRIPTION[1]	PILOT RATING
No vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of *primary task* is *affected* or *tasks can only* be *done* with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration level.	10

[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 3. Vibration Rating Scale

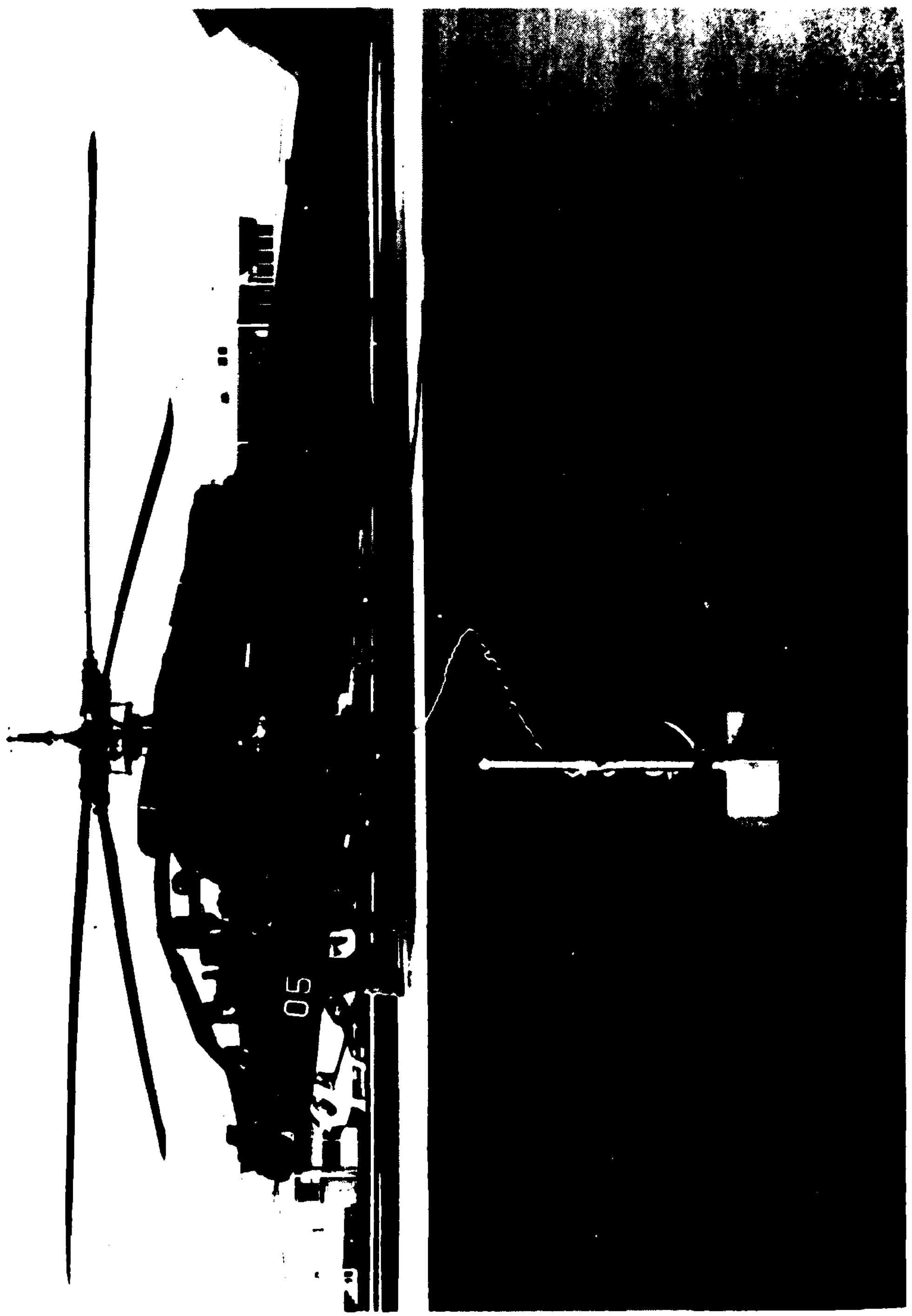


Photo 1. Trailing Bomb (foreground)

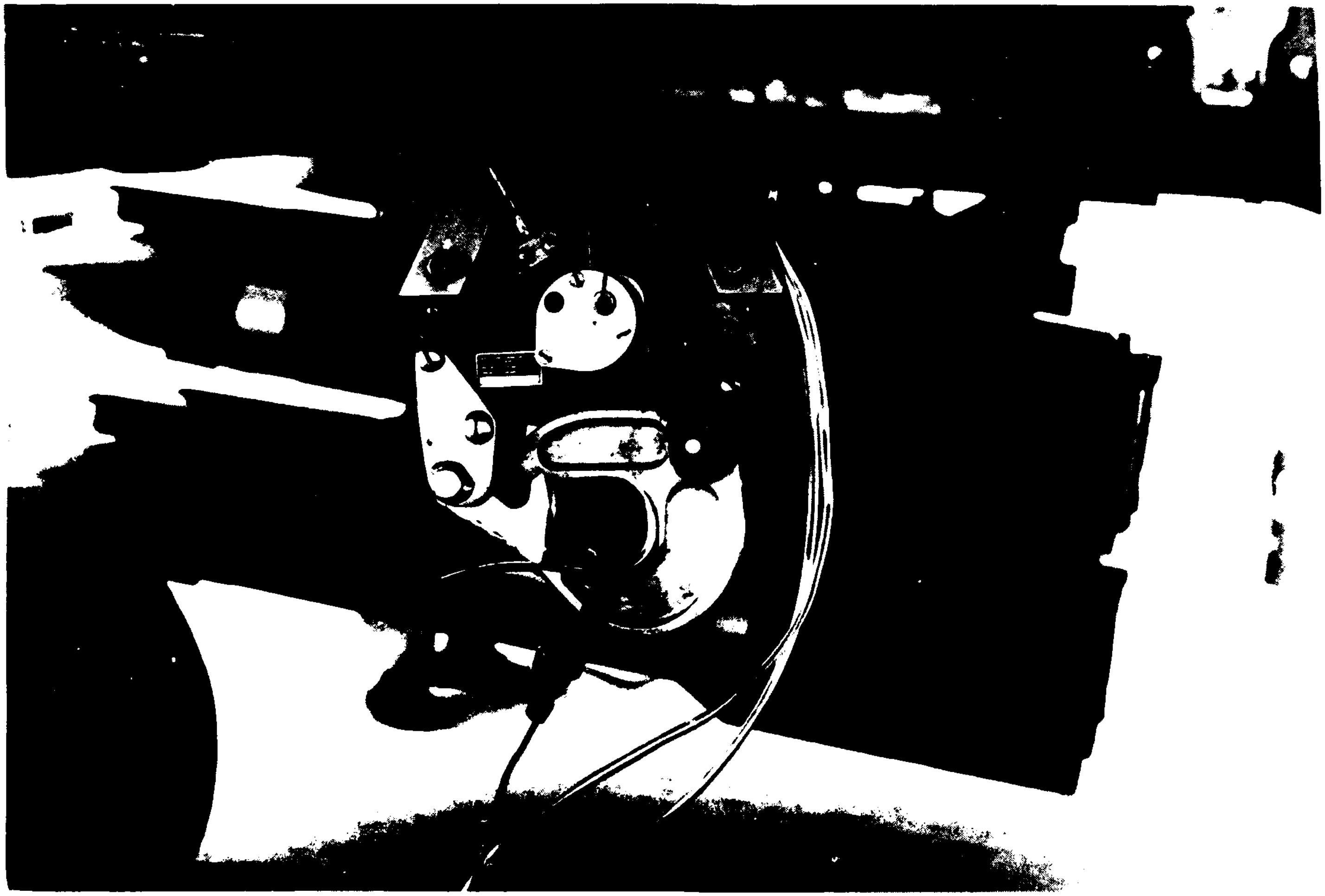


Photo 2. Cargo Hook Installation for Attachment of Trailing Bomb

a. Deficiency - A defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued; or indicates improper design or other cause of failure of an item or part, which seriously impairs the equipment's operational capability.

b. Shortcoming - An imperfection or malfunction occurring during the life cycle of equipment which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the useability of the material or end product.

APPENDIX E. TEST DATA

INDEX

FIGURE I
NONDIMENSIONAL HOVER PERFORMANCE
YAH-64 USA S/N 77-23258

WHEEL HEIGHT = 100 FEET

SYMBOL	AVG ROTOR SPEED (RPM)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)
▼	286	-50	13.0
⊙	289	240	13.5

NOTES: 1. FREEFLIGHT HOVER TECHNIQUE
2. 8-HELLFIRE CONFIGURATION
3. WINDS 3 KNOTS OR LESS

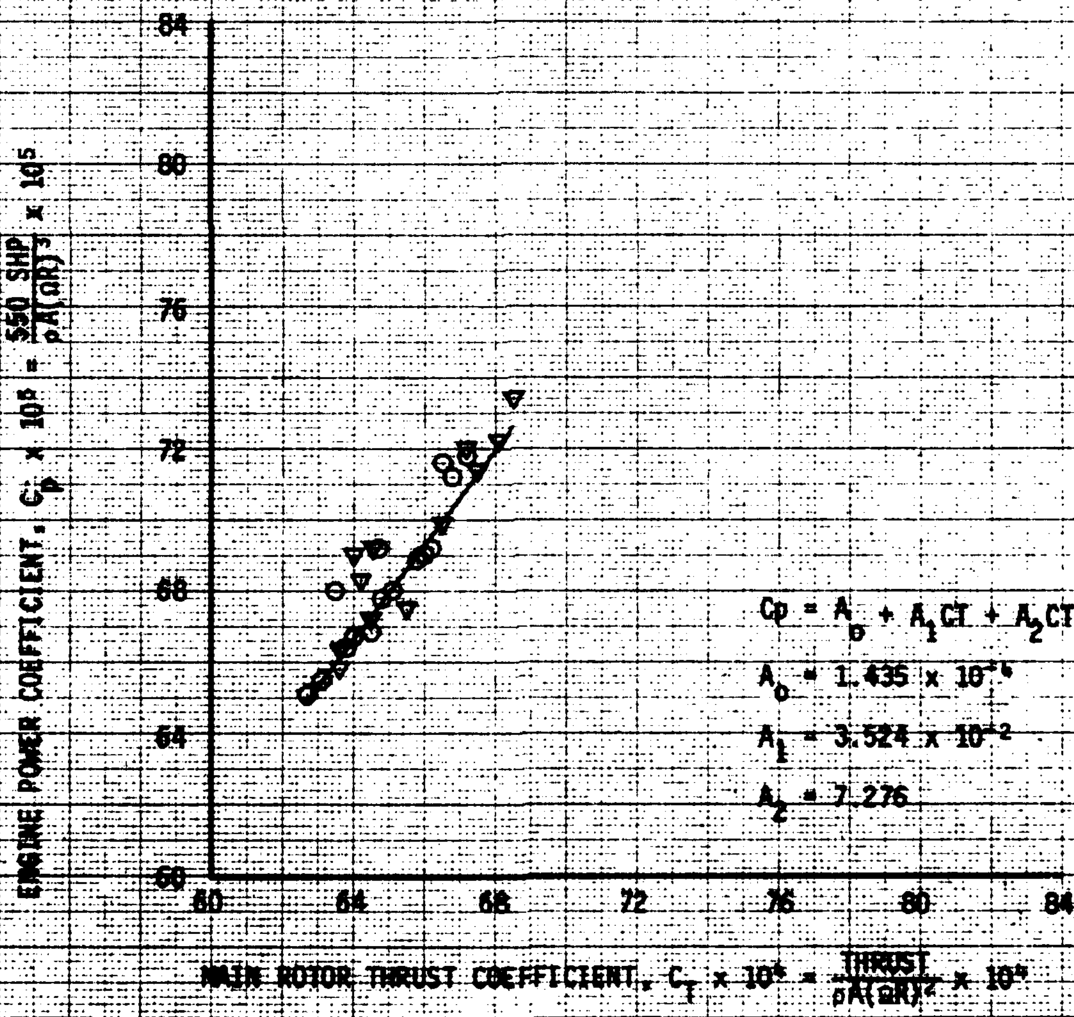


FIGURE 2
NONDIMENSIONAL VERTICAL CLIMB PERFORMANCE
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (^{0}C)	AVG ROTOR SPEED (RPM)
⊙	14,600	206.8(AFT)	0.5(RT)	550	14.0	289
⊡	14,680	206.3(AFT)	0.5(RT)	-80	12.0	286

NOTE: 1. 8-HELLFIRE CONFIGURATION
2. WINDS LESS THAN 3 KNOTS

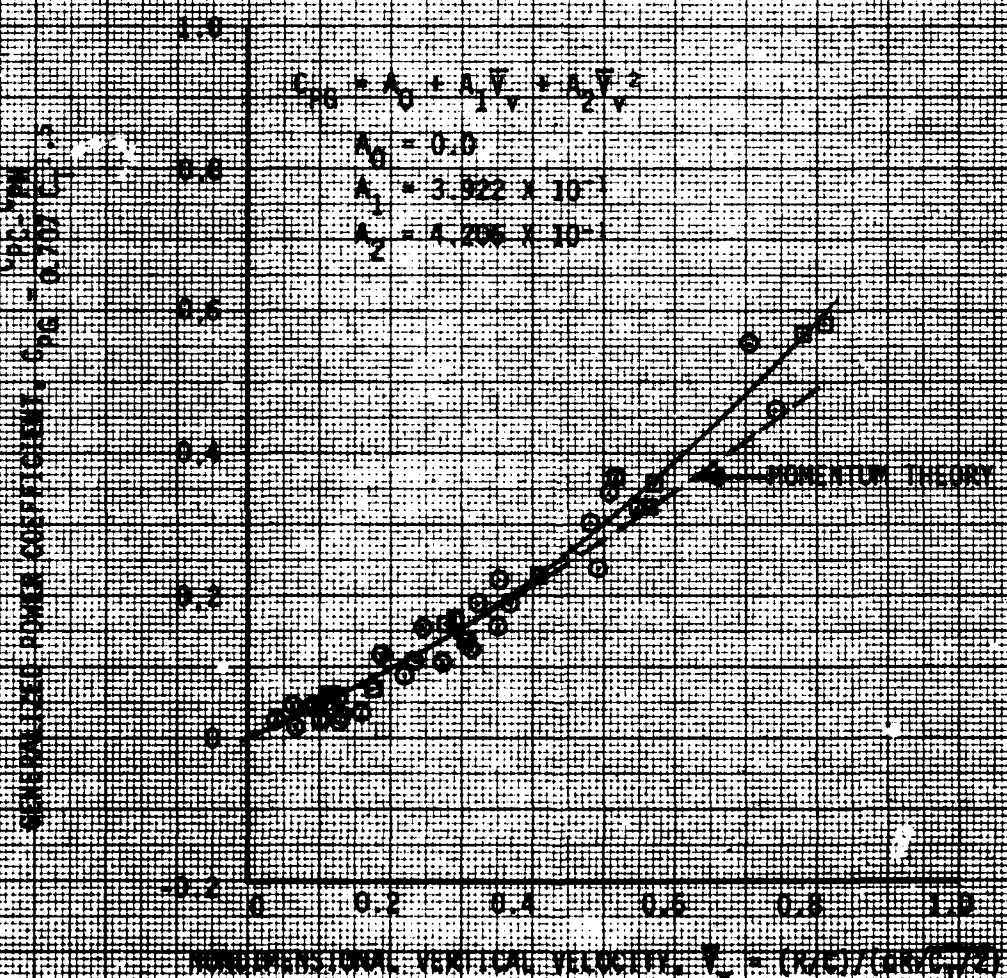


FIGURE 3
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 77-23258

NOTES:
1. ROTORS STATIC
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS
4. COPILOT/GUNNER CYCLIC CONTROL EXTENDED
5. TRIM FEEL ON
6. TOTAL LONGITUDINAL CONTROL TRAVEL = 10.4 INCHES

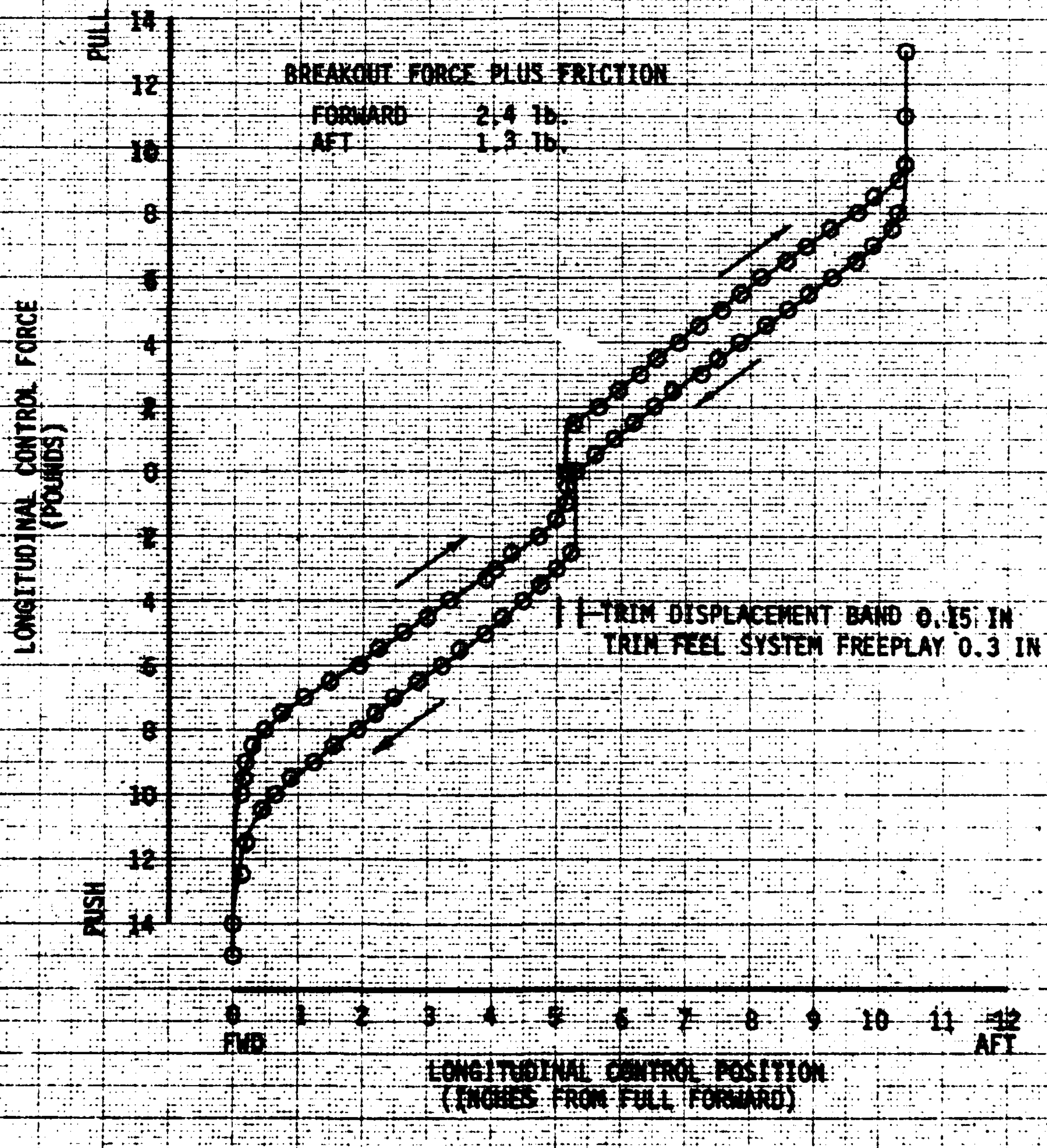
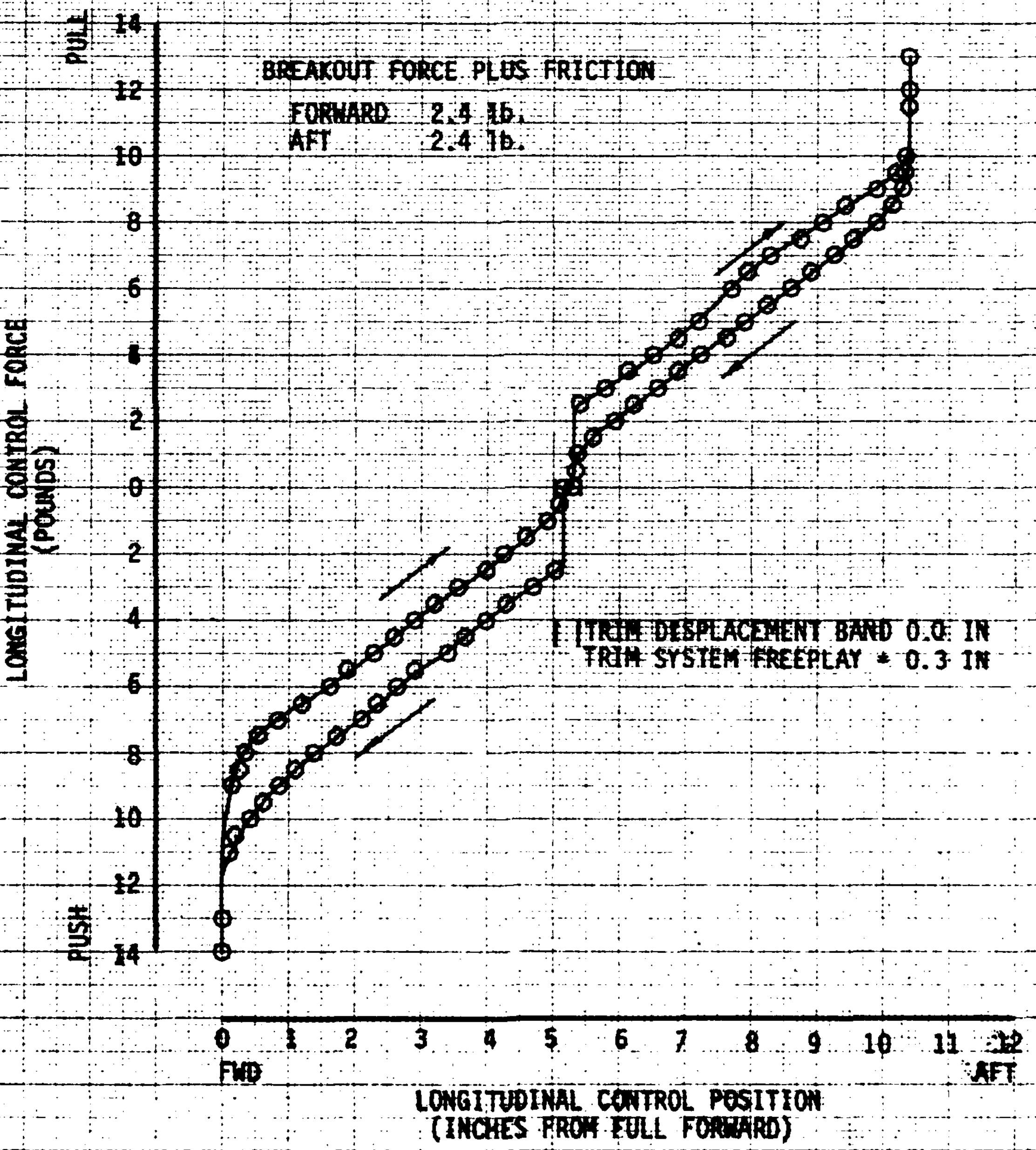
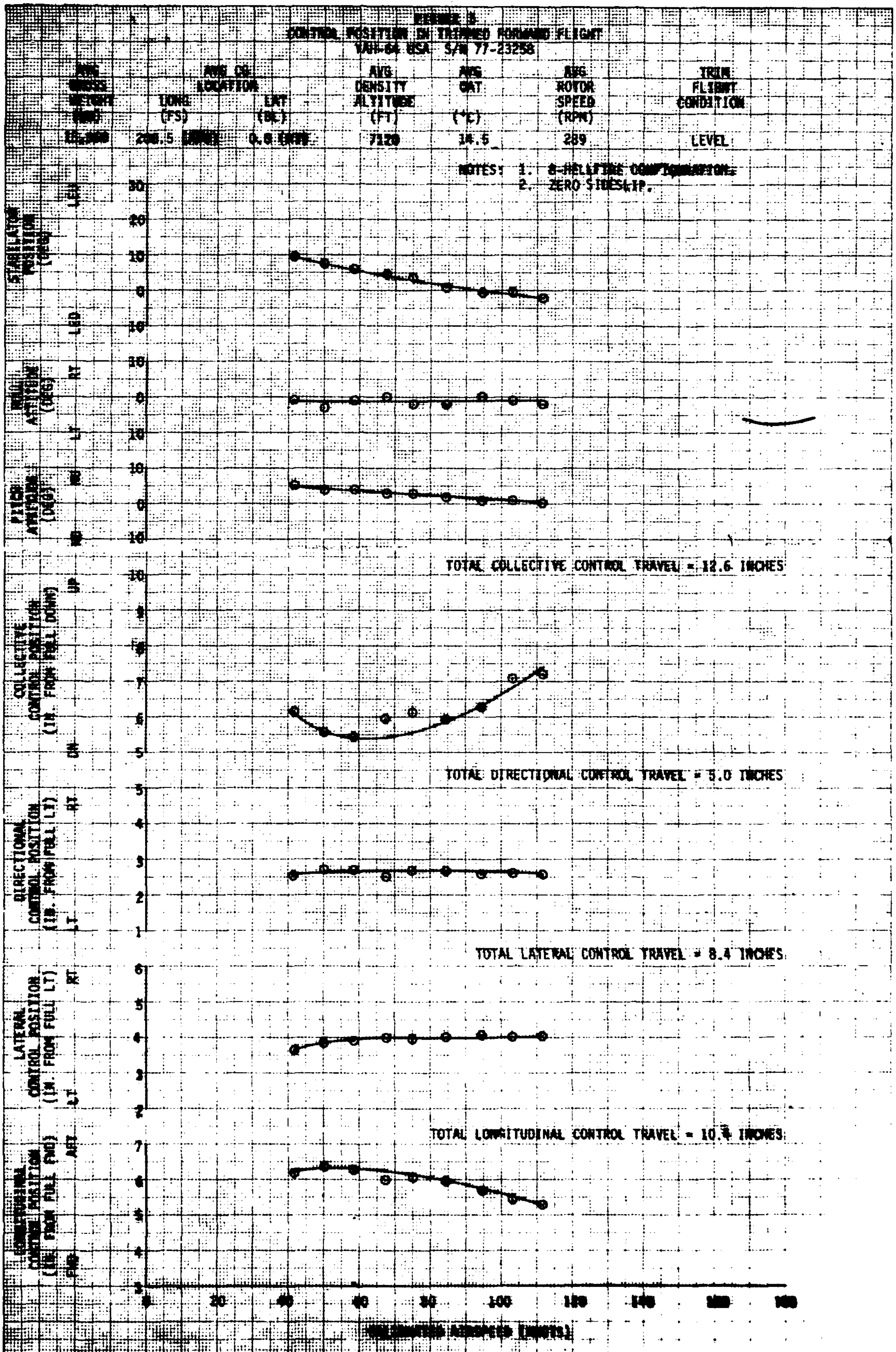
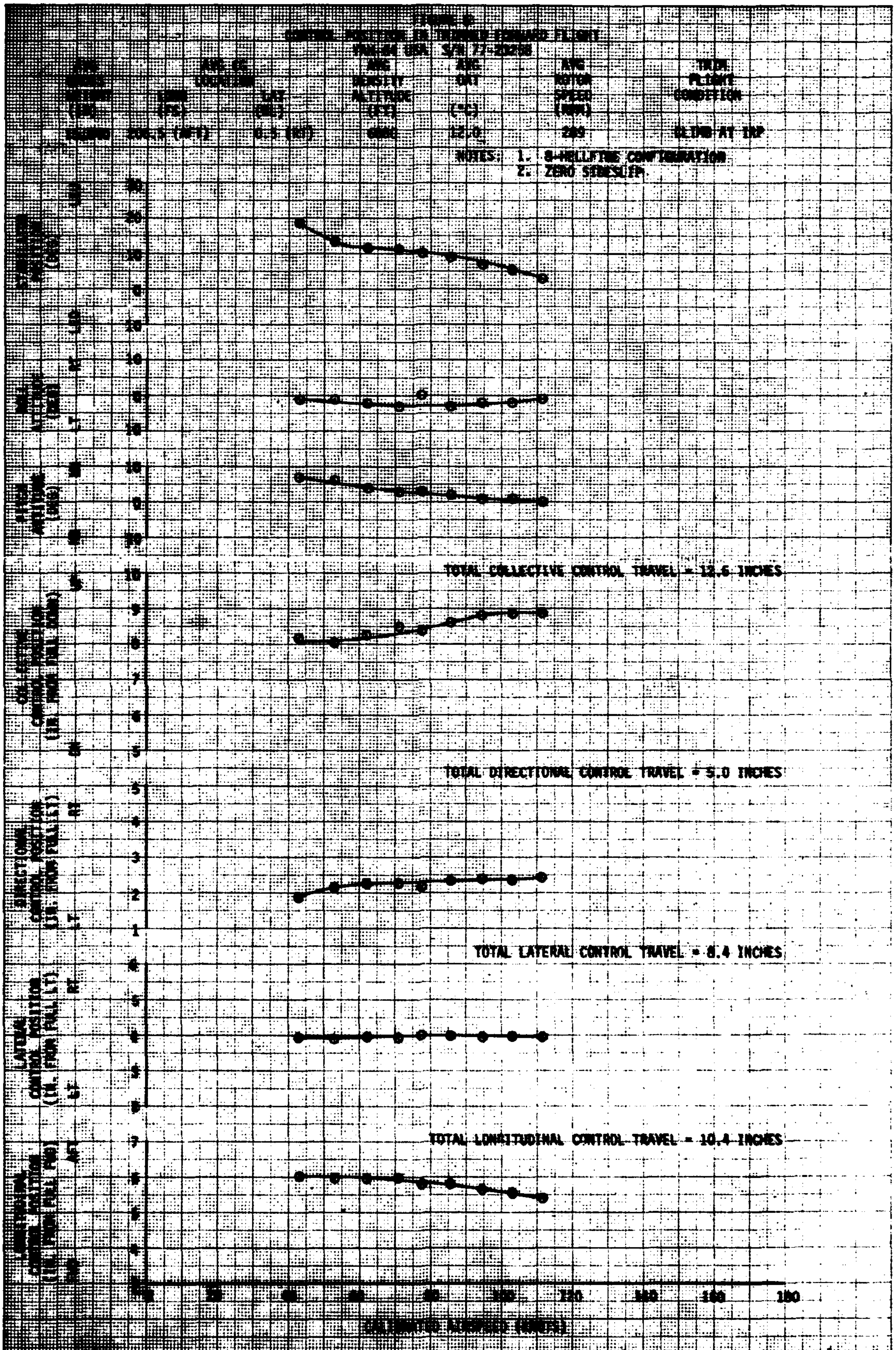


FIGURE 4
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 77-23258

NOTES: 1. ROTORS STATIC
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS
4. COPILOT/GUNNER CYCLIC CONTROL RETRACTED
5. TRIM FEEL ON
6. TOTAL LONGITUDINAL CONTROL TRAVEL = 10.4 INCHES




CONTROL POSITION IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
TRIM FLIGHT CONDITION
7120
14.5
289
LEVEL
NOTES:
2. ZERO SIDESLIP.
TOTAL COLLECTIVE CONTROL TRAVEL = 12.6 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.0 INCHES
TOTAL LATERAL CONTROL TRAVEL = 8.4 INCHES
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LT)
LATERAL CONTROL POSITION (IN. FROM FULL LT)
COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN)
PITCH ATTITUDE (DEG)
LEVEL


AVG
AVG CG LOCATION
AVG DENSITY ALTITUDE
AVG OAT
AVG ROTOR SPEED
TRIM FLIGHT CONDITION
12.0
NOTES: 1. 8-HELLFIRE CONFIGURATION
2. ZERO SIDESLIP
TOTAL COLLECTIVE CONTROL TRAVEL = 12.6 INCHES
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.0 INCHES
TOTAL LATERAL CONTROL TRAVEL = 8.4 INCHES
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.4 INCHES

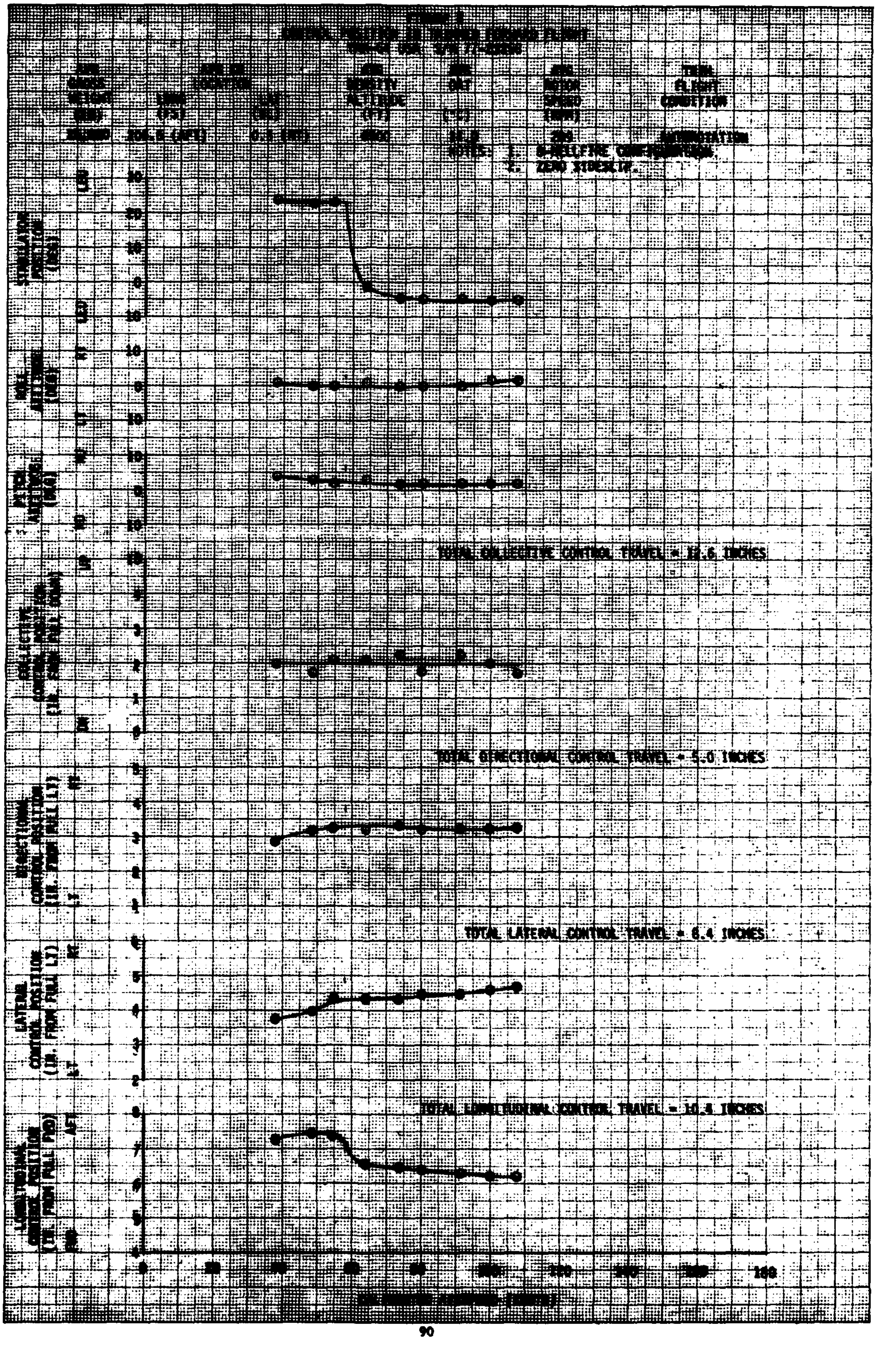

NOTES: 1.
2. ZERO SIDESLIP.
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.0 INCHES
TOTAL LATERAL CONTROL TRAVEL = 6.4 INCHES
LATERAL CONTROL POSITION (IN. FROM FULL LT)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LT)

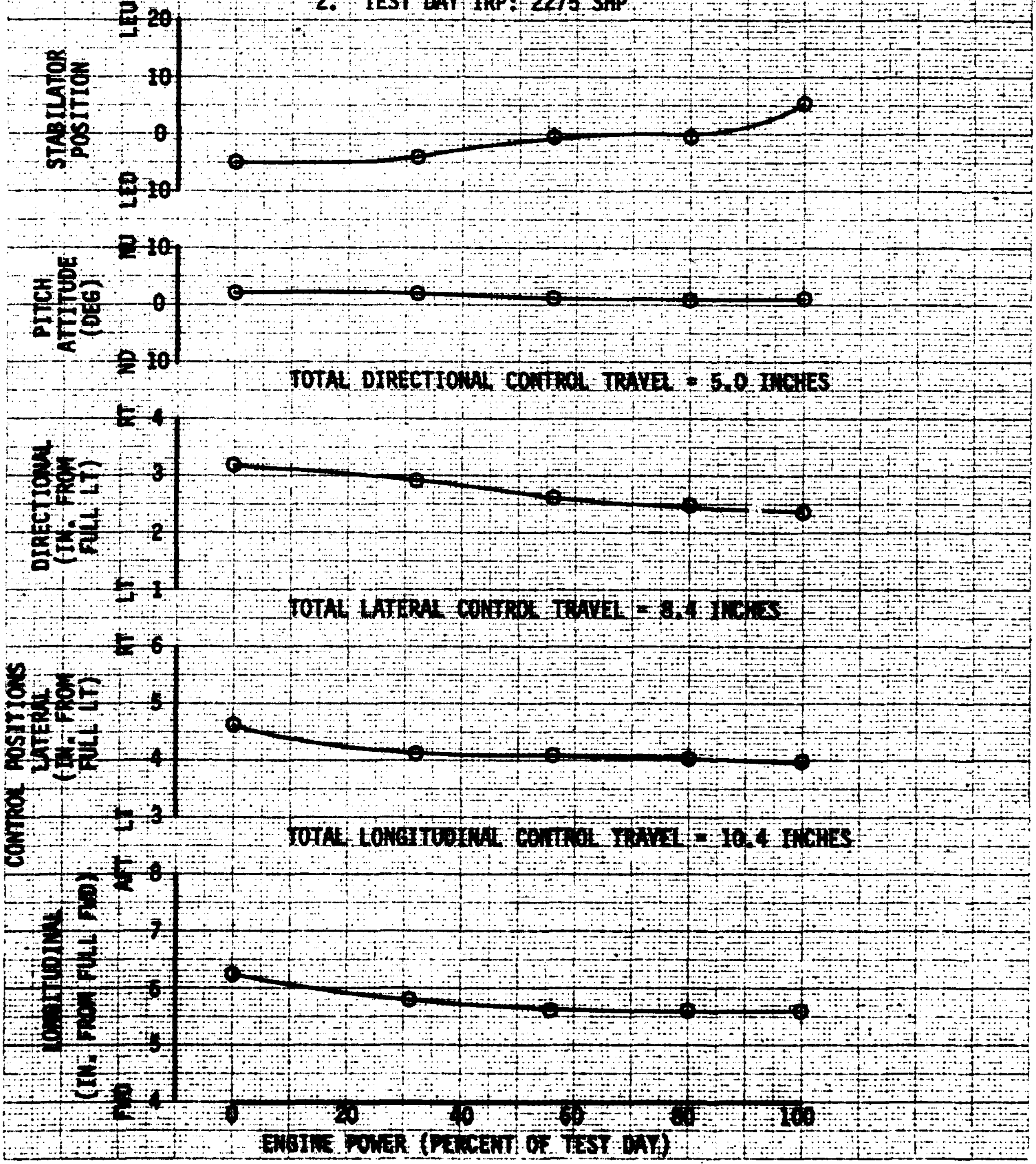
FIGURE 8

TRIMMED CONTROL POSITION VARIATION WITH ENGINE POWER

YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KT)
14,500	206.5(AFT)	0.5(RT)	6020	13.0	289	100

NOTES: 1. ZERO SIDESLIP, DASE ON, 8-HELLFIRE CONFIGURATION
2. TEST DAY IRP: 2275 SHP



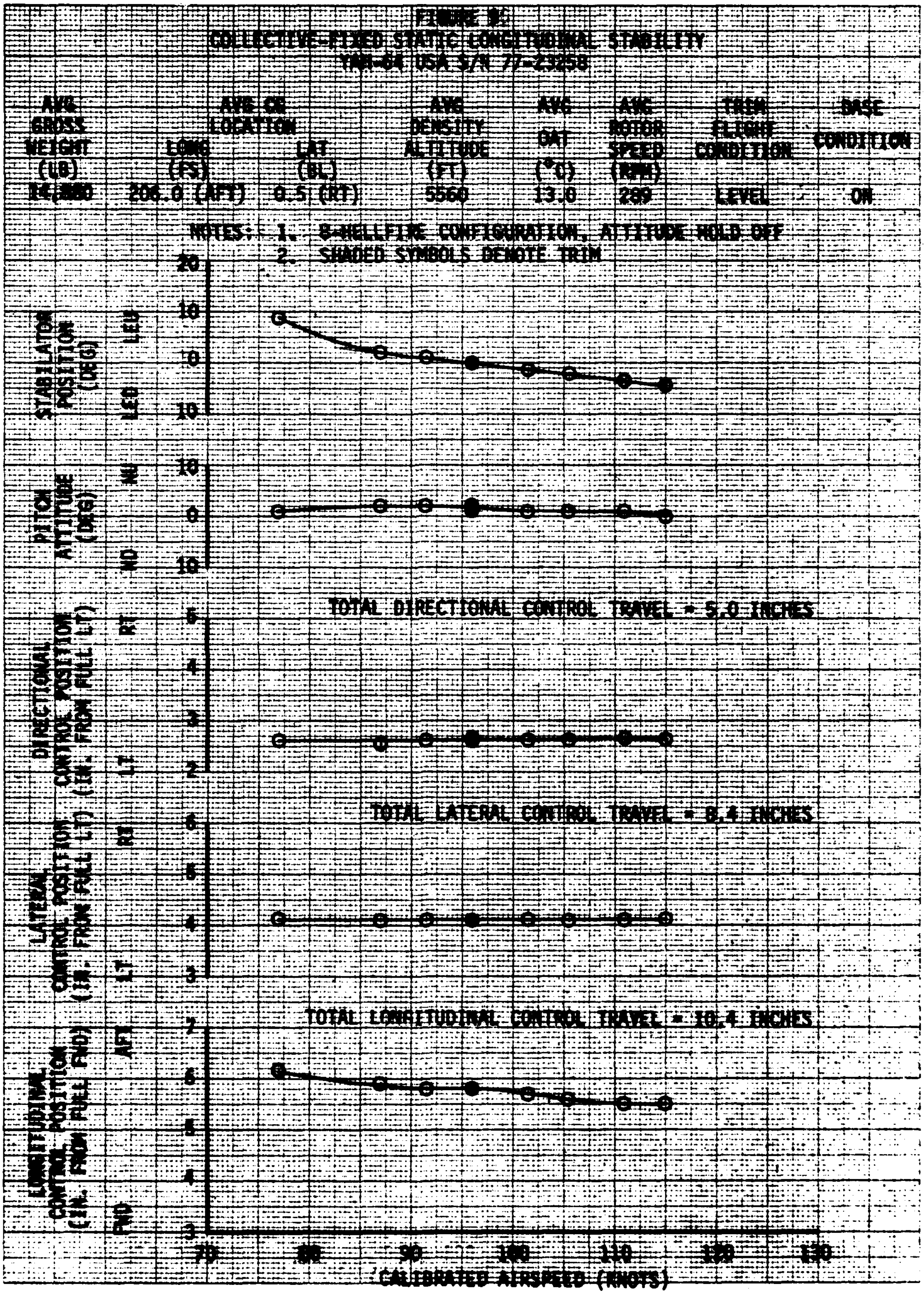
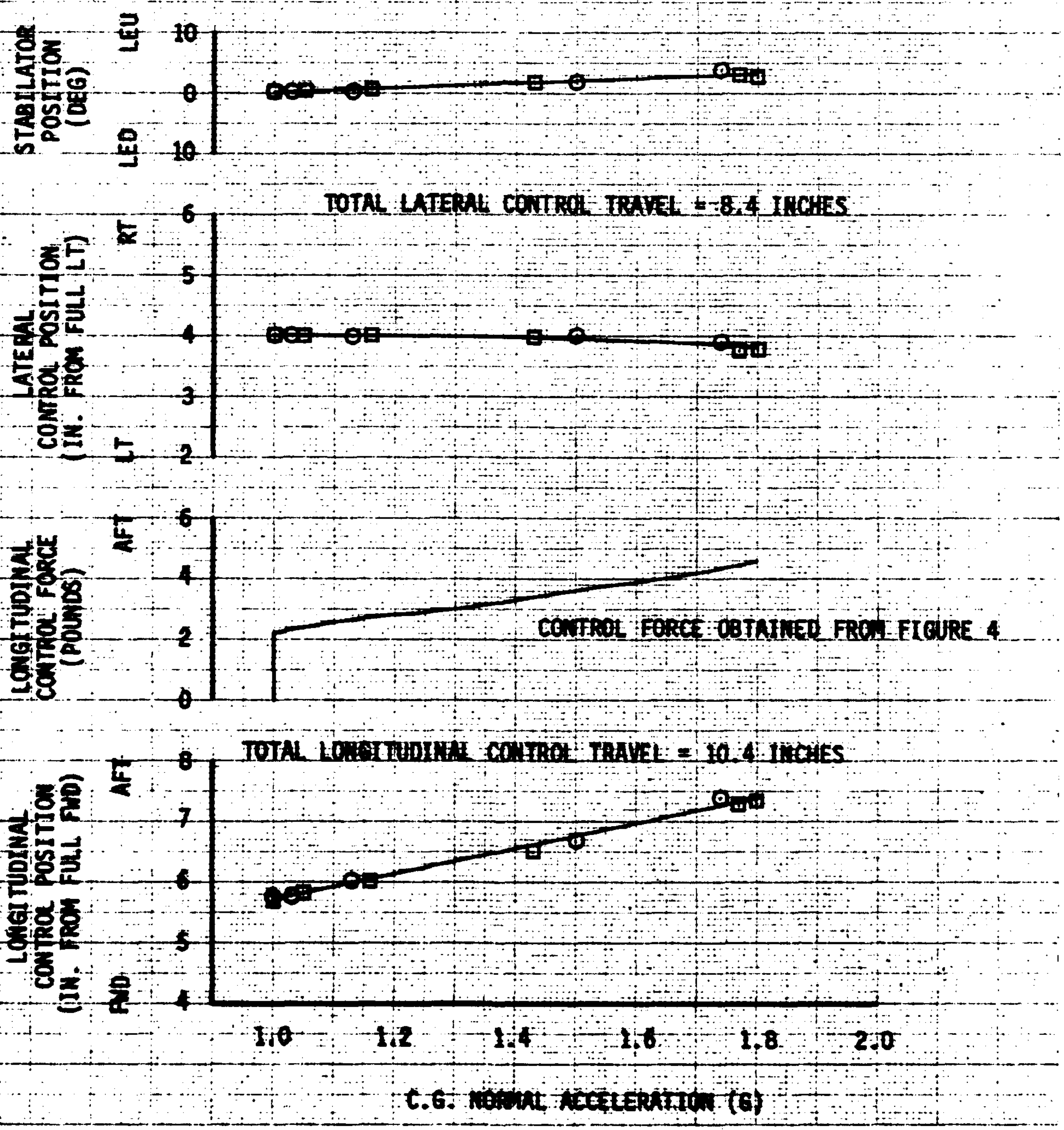


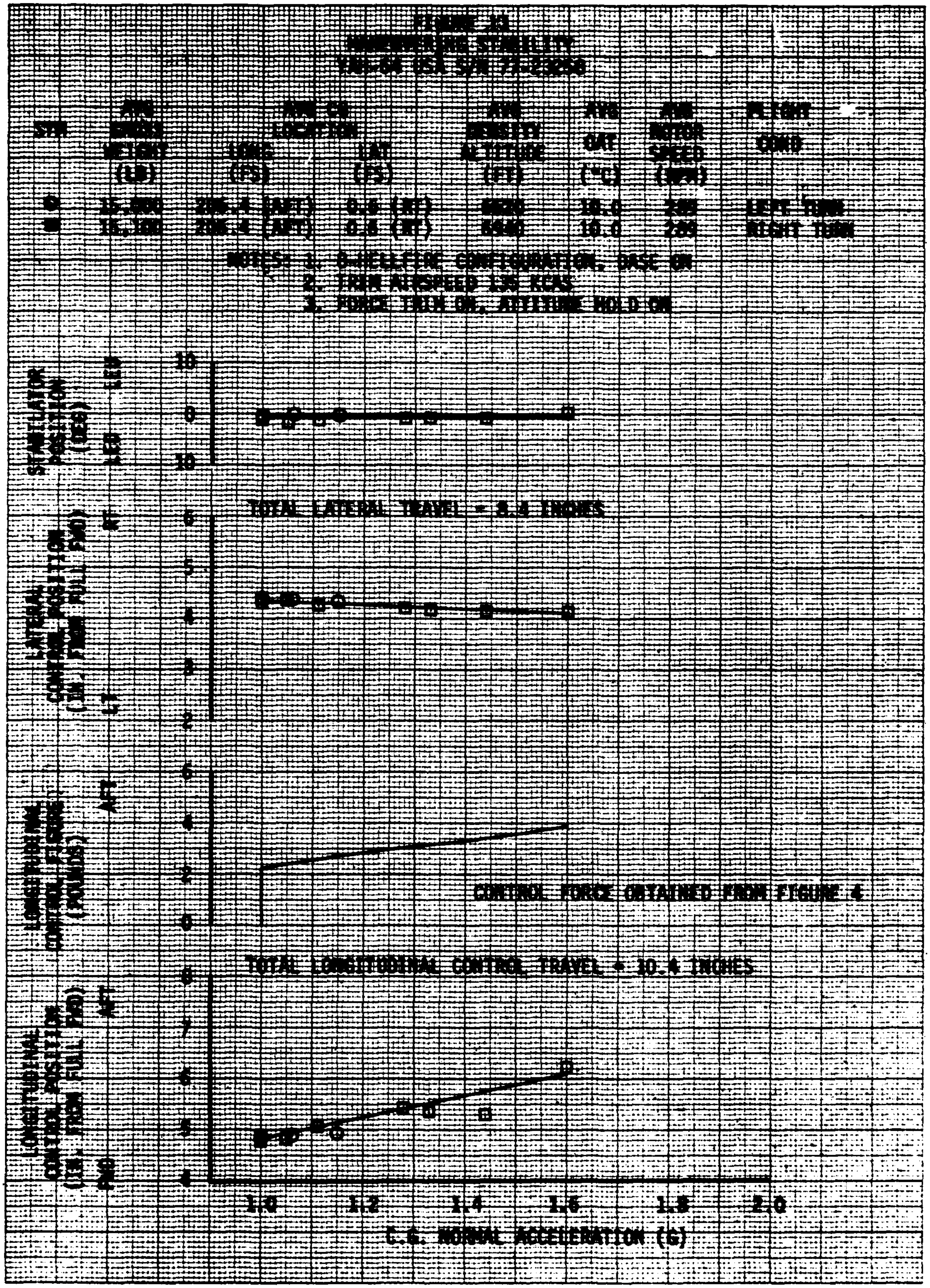
FIGURE
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
TRIM FLIGHT CONDITION
DASE CONDITION
206.0 (AFT)
0.5 (RT)
5560
13.0
289
LEVEL
ON
NOTES: 1. 8-HELLFIRE CONFIGURATION, ATTITUDE HOLD OFF
2. SHADED SYMBOLS DENOTE TRIM
STABILATOR POSITION (DEG)
LED
TED
PITCH ATTITUDE (DEG)
NU
ND
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.0 INCHES
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LT)
RT
LT
TOTAL LATERAL CONTROL TRAVEL = 8.4 INCHES
LATERAL CONTROL POSITION (IN. FROM FULL LT)
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.4 INCHES
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
AFT
FWD
20
10
0
10
10
0
10
5
4
3
2
6
5
4
3
7
6
5
4
3
70
80
90
100
110
120
130
CALIBRATED AIRSPEED (KNOTS)

FIGURE 10
MANEUVERING STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT COND
⊙	15,140	205.5 (AFT)	0.6 (RT)	4960	13.0	289	LEFT TURN
⊡	14,980	205.9 (AFT)	0.6 (RT)	5220	13.0	289	RIGHT TURN

NOTES: 1. 8-HELLFIRE CONFIGURATION, DASE ON
2. TRIM AIRSPEED 101 KCAS
3. FORCE TRIM ON, ATTITUDE HOLD OFF




3. FORCE TRIM ON, ATTITUDE HOLD ON
TOTAL LATERAL TRAVEL
CONTROL FORCE OBTAINED FROM FIGURE 4
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.4 INCHES
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
1.0
1.2
1.4
1.6
1.8
2.0
C.G. NORMAL ACCELERATION (G)

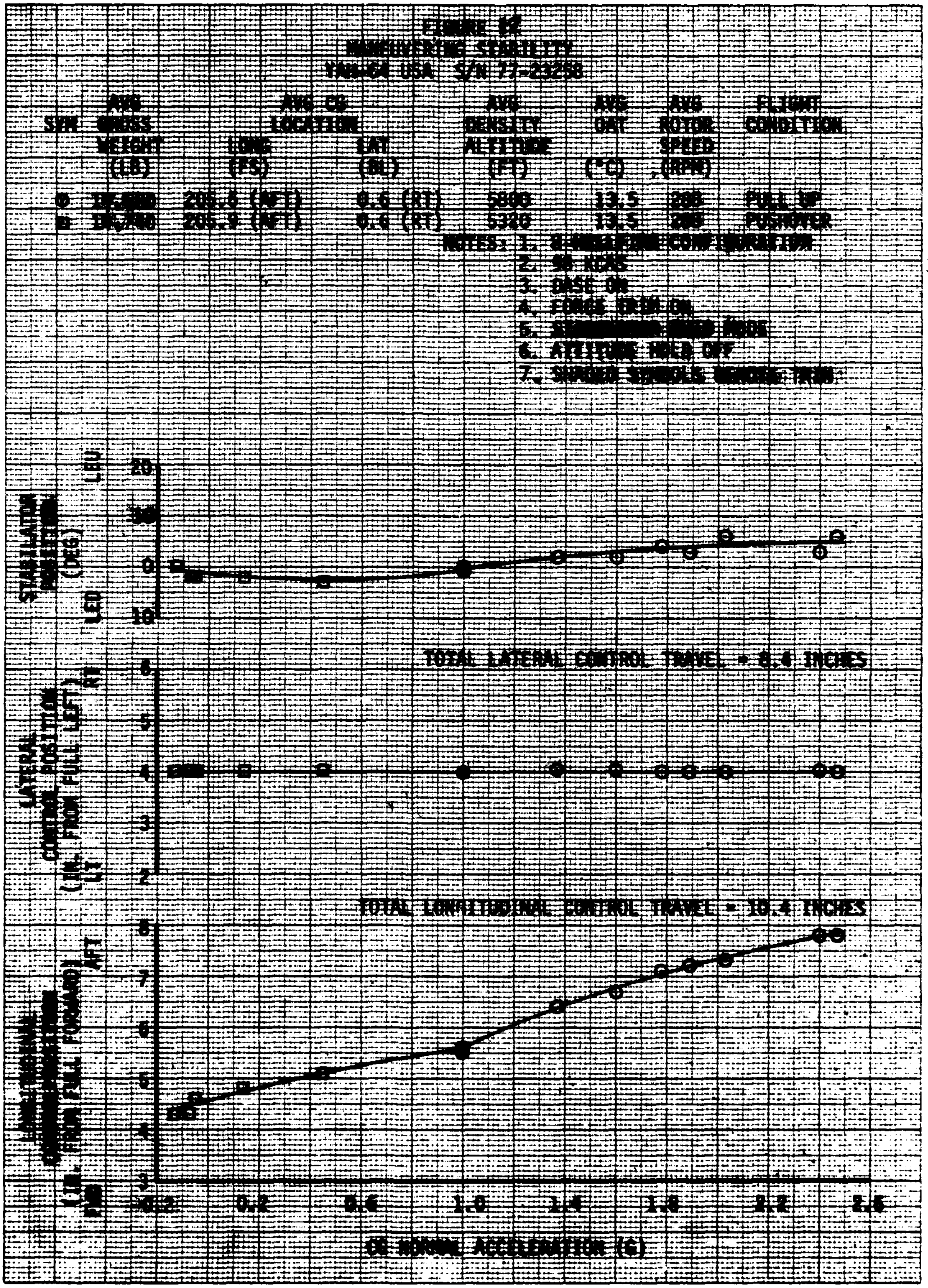

MANEUVERING STABILITY
USA S/N 77-23258
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
205.6 (AFT) 0.6 (RT) 5800 13.5 PULL UP
205.9 (AFT) 0.6 (RT) 5320 13.5 PUSHOVER
2.
3. DASE ON
4. FORCE TRIM ON
6. ATTITUDE HOLD OFF
STABILATOR POSITION (DEG)
TOTAL LATERAL CONTROL TRAVEL
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.4 INCHES
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
FROM FULL FORWARD) AFT
0.2
0.6
1.0
1.4
1.8
2.2
2.6
NORMAL ACCELERATION (G)

FIGURE 13
MANEUVERING STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
○	14,740	206.5 (AFT)	0.6 (RT)	6060	11.0	289	PULL UP
□	14,480	206.5 (AFT)	0.6 (RT)	6240	11.0	289	PUSHOVER

NOTES:
1. 8-HELLFIRE CONFIGURATION
2. 134 KCAS
3. DASE ON
4. FORCE TRIM ON
5. STABILATOR [illegible]
6. ATTITUDE HOLD OFF
7. SHADED SYMBOLS DENOTE TRIM

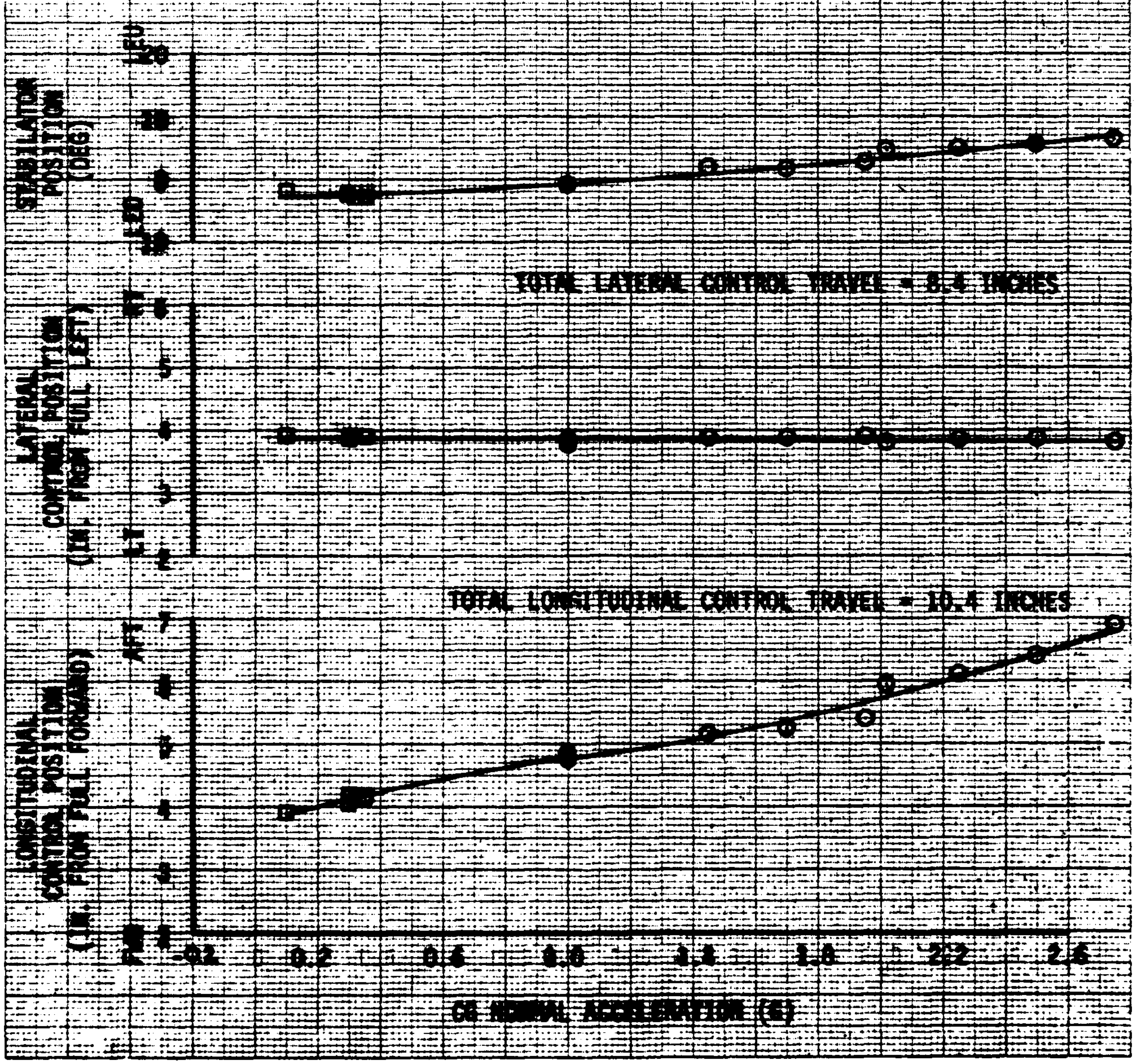


FIGURE 14
LONGITUDINAL SHORT TERM RESPONSE
YAH-64 USA S/N 77-23258

AVG. GROSS WEIGHT (LB)	AVG. C.G. LOCATION LONG (FS)	AVG. C.G. LOCATION LAT. (BL)	AVG. DENSITY ALTITUDE (FT)	AVG. OAT (C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,660	206.0 (AFT)	0.5 (RT)	5,460	15.0	289	LEVEL	ON	124

NOTE: AIRCRAFT CONFIGURATION: 8 HELLFIRE

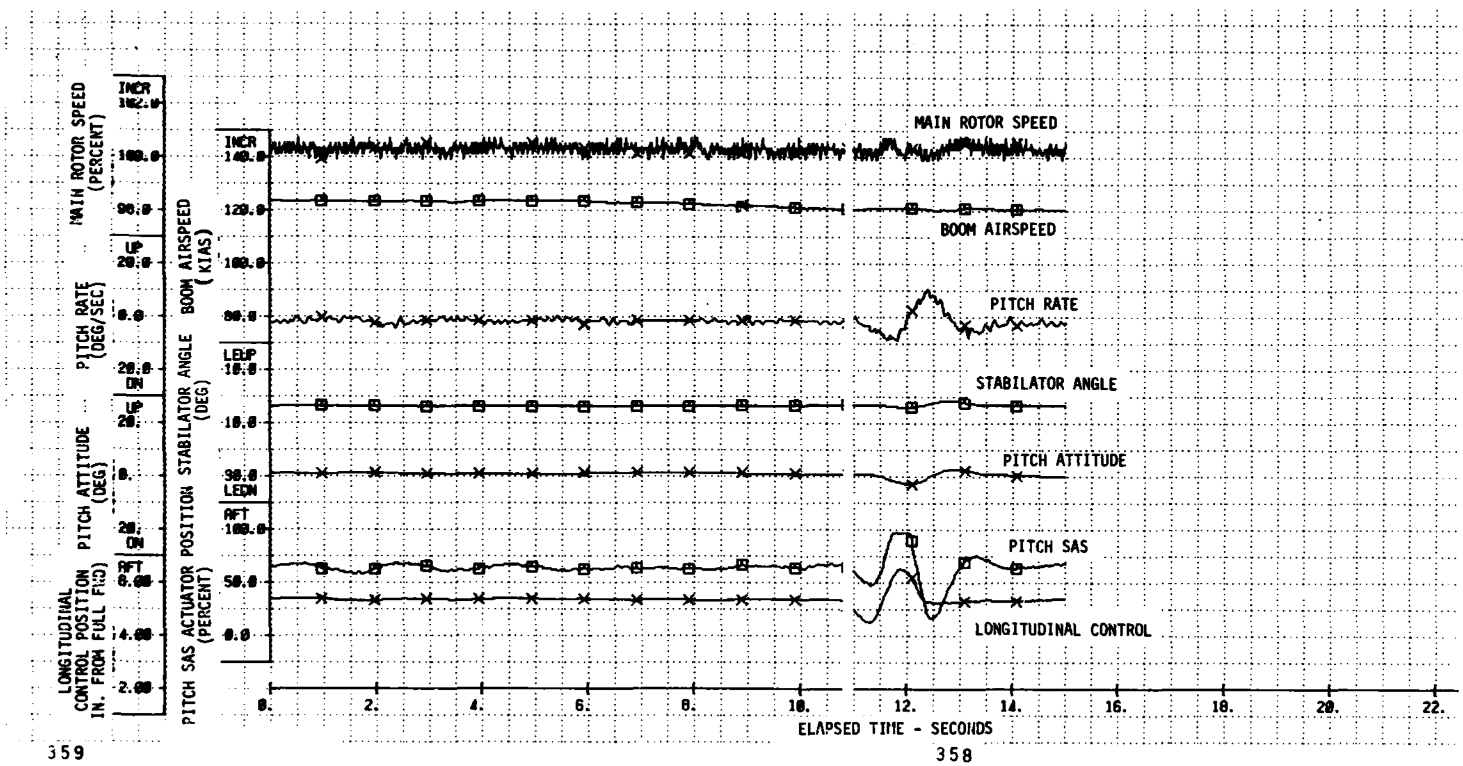


FIGURE 15
LONGITUDINAL LONG TERM RESPONSE
YAH-64 USA S/N 77-23258

AVG. GROSS WEIGHT (LB)	AVG. C.G. LOCATION LONG (FS)	AVG. C.G. LOCATION LAT. (BL)	AVG. DENSITY ALTITUDE (FT)	AVG. OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,680	206.0 (AFT)	0.5 (RT)	5,400	14.5	289	LEVEL	ON	98

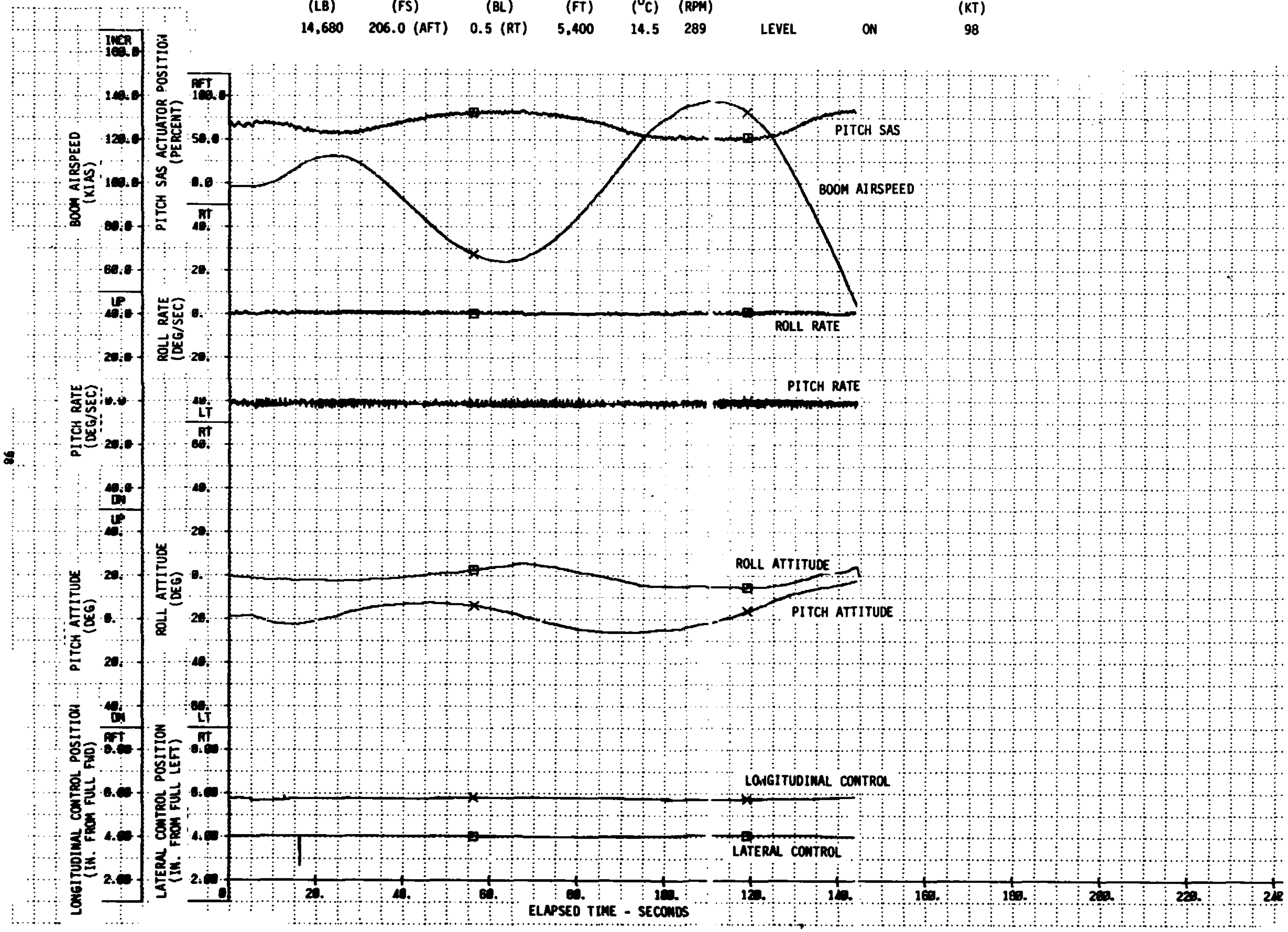
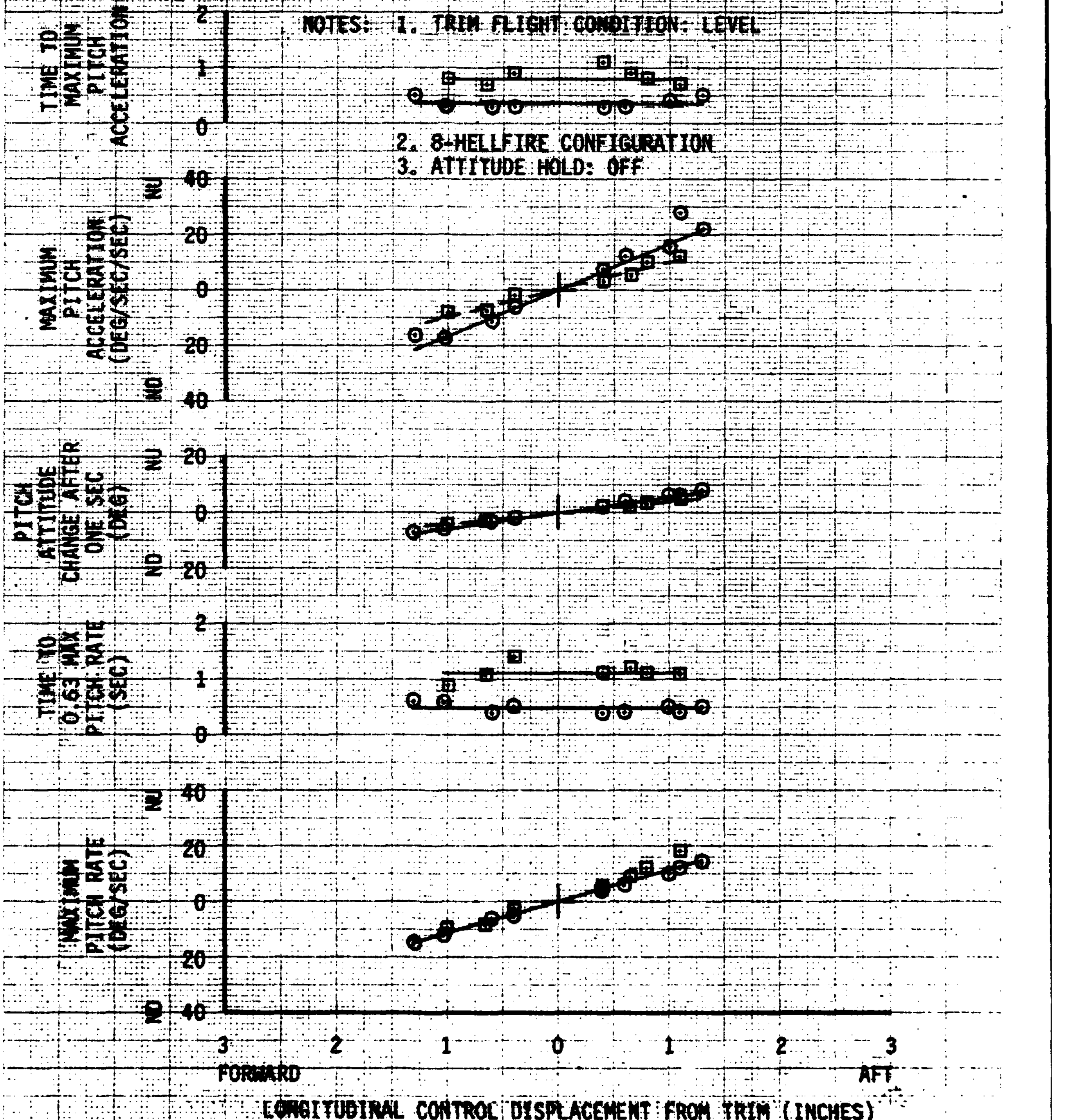


FIGURE 16
LONGITUDINAL CONTROLLABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	DASE CONDITION
⊙	15,040	206.7 (AFT)	0.5 (RT)	5160	10.5	289	100	ON
⊡	14,840	206.7 (AFT)	0.5 (RT)	5480	10.5	289	100	OFF



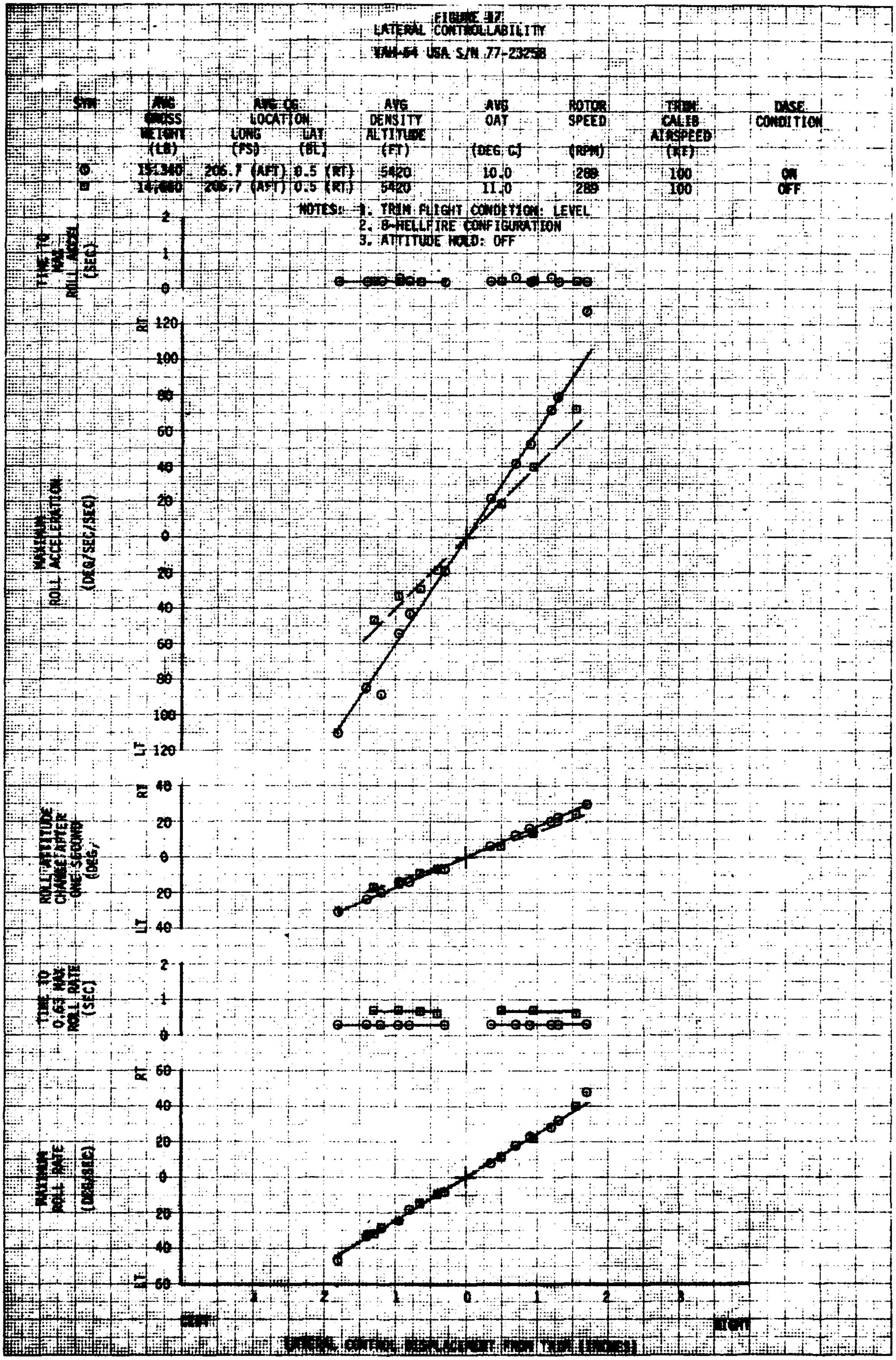


FIGURE 37
LATERAL CONTROLLABILITY
YAH-64 USA S/N 77-23258
SYM
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM CALIB AIRSPEED (KT)
DASE CONDITION
15,340 206.7 (AFT) 0.5 (RT) 5420 10.0 289 100 ON
206.7 (AFT) 0.5 (RT) 5420 11.0 289 100 OFF
NOTES: 1. TRIM FLIGHT CONDITION: LEVEL
2. 8-HELLFIRE CONFIGURATION
3. ATTITUDE HOLD: OFF
TIME TO MAX ROLL ACCEL (SEC)
MAXIMUM ROLL ACCELERATION (DEG/SEC/SEC)
ROLL ATTITUDE CHANGE AFTER ONE SECOND (DEG)
TIME TO 0.63 MAX ROLL RATE (SEC)
MAXIMUM ROLL RATE (DEG/SEC)
RT
LT
LEFT
RIGHT
LATERAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

FIGURE 10
AFT LONGITUDINAL STEP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (° C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED
15,540	206.6(AFT)	0.5(RT)	5040	11.0	289	LEVEL	ON	98

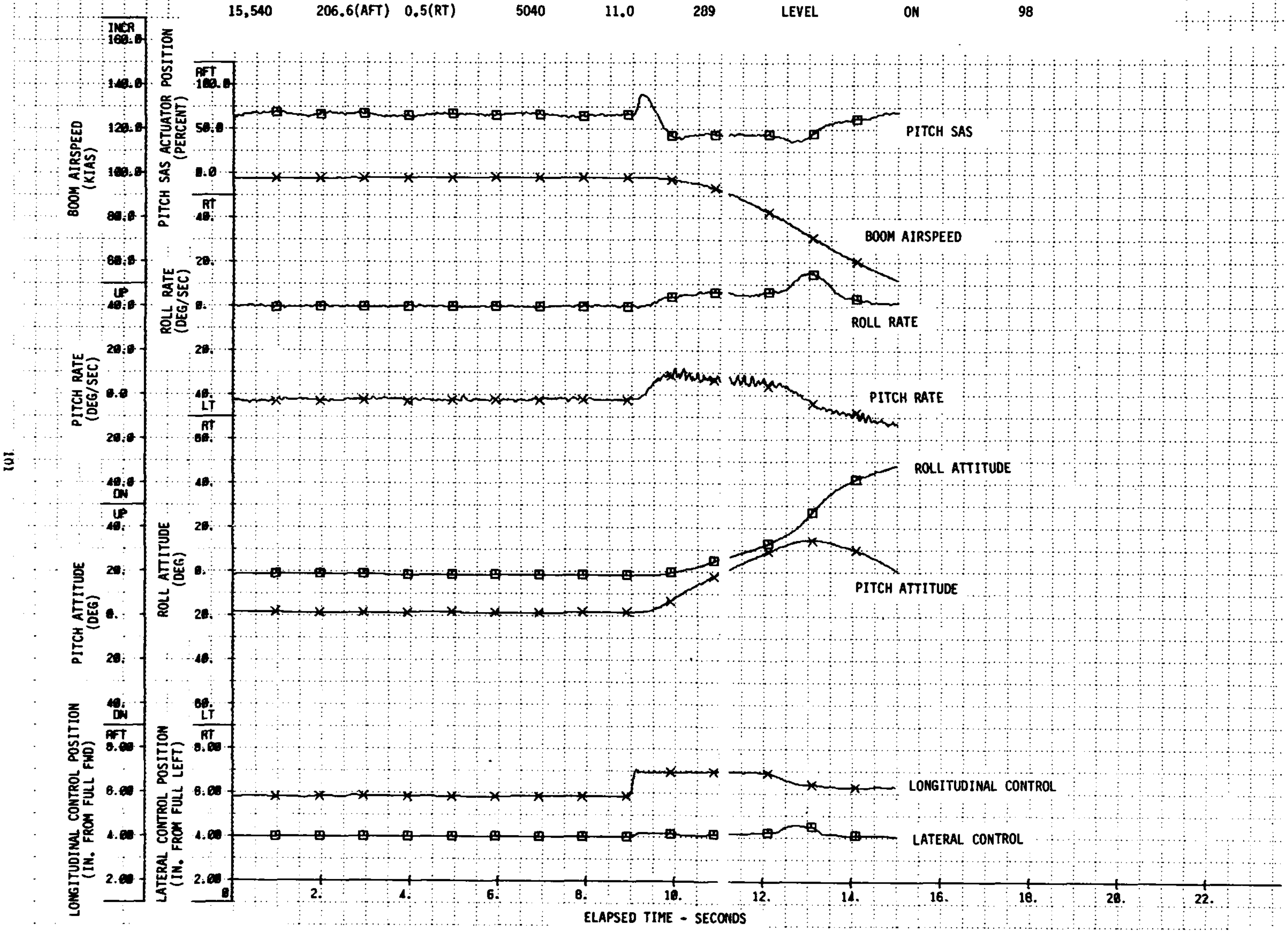


FIGURE 19
LEFT LATERAL STEP
YAH-64 USA S/N 77-23258

AVG. GROSS WEIGHT (LB)	AVG. C.G. LOCATION LONG (FS)	AVG. C.G. LOCATION LAT. (BL)	AVG. DENSITY ALTITUDE (FT)	AVG. OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,660	206.6 (AFT)	0.5 (RT)	5400	10.5	289	LEVEL	ON	98

NOTE: AIRCRAFT CONFIGURATION: 8 HELLFIRE

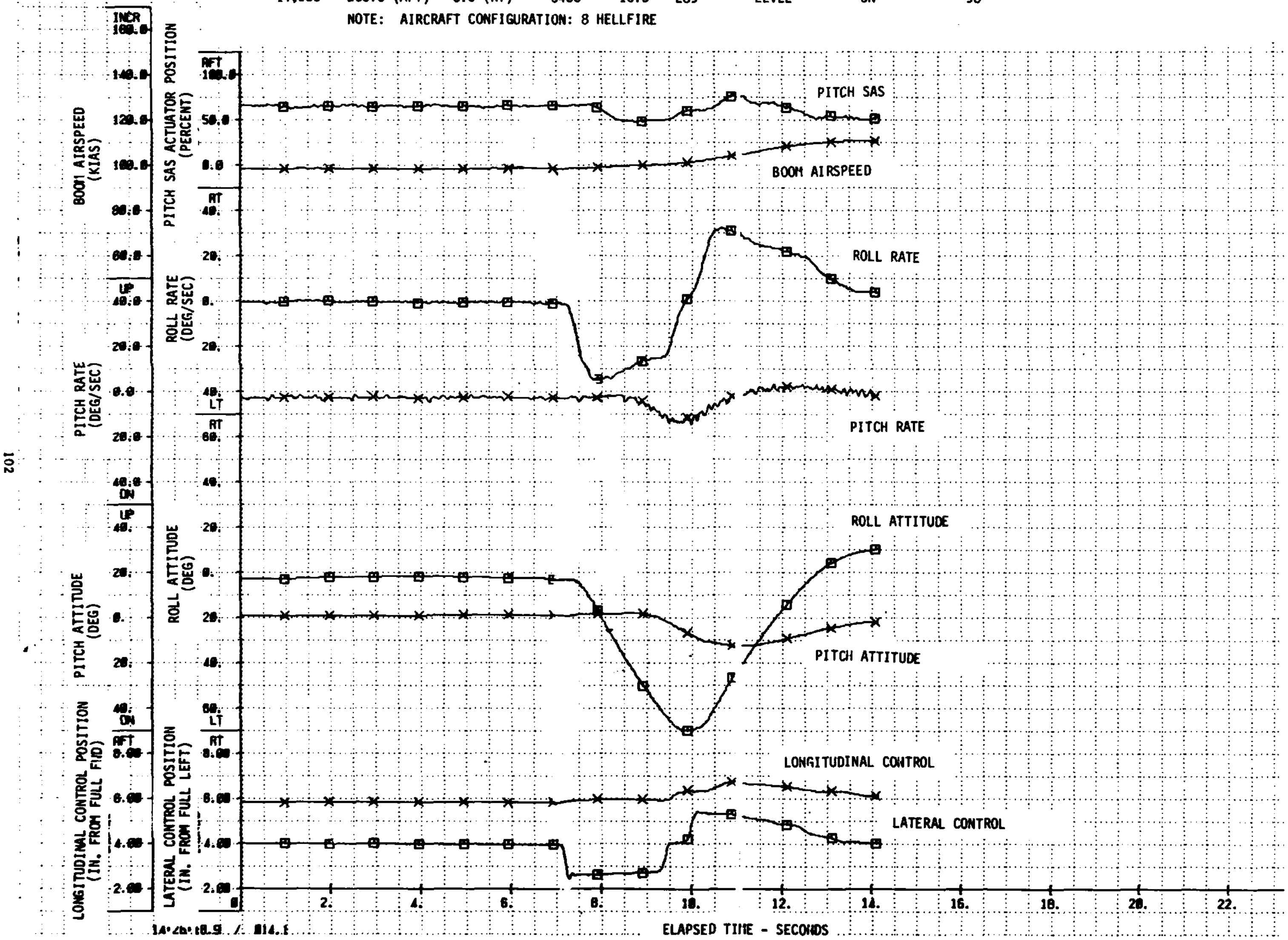


FIGURE 20
AFT LONGITUDINAL STEP
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (° C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED
15,480	206.6(AFT)	0.5(RT)	5060	11.0	289	LEVEL	OFF	99

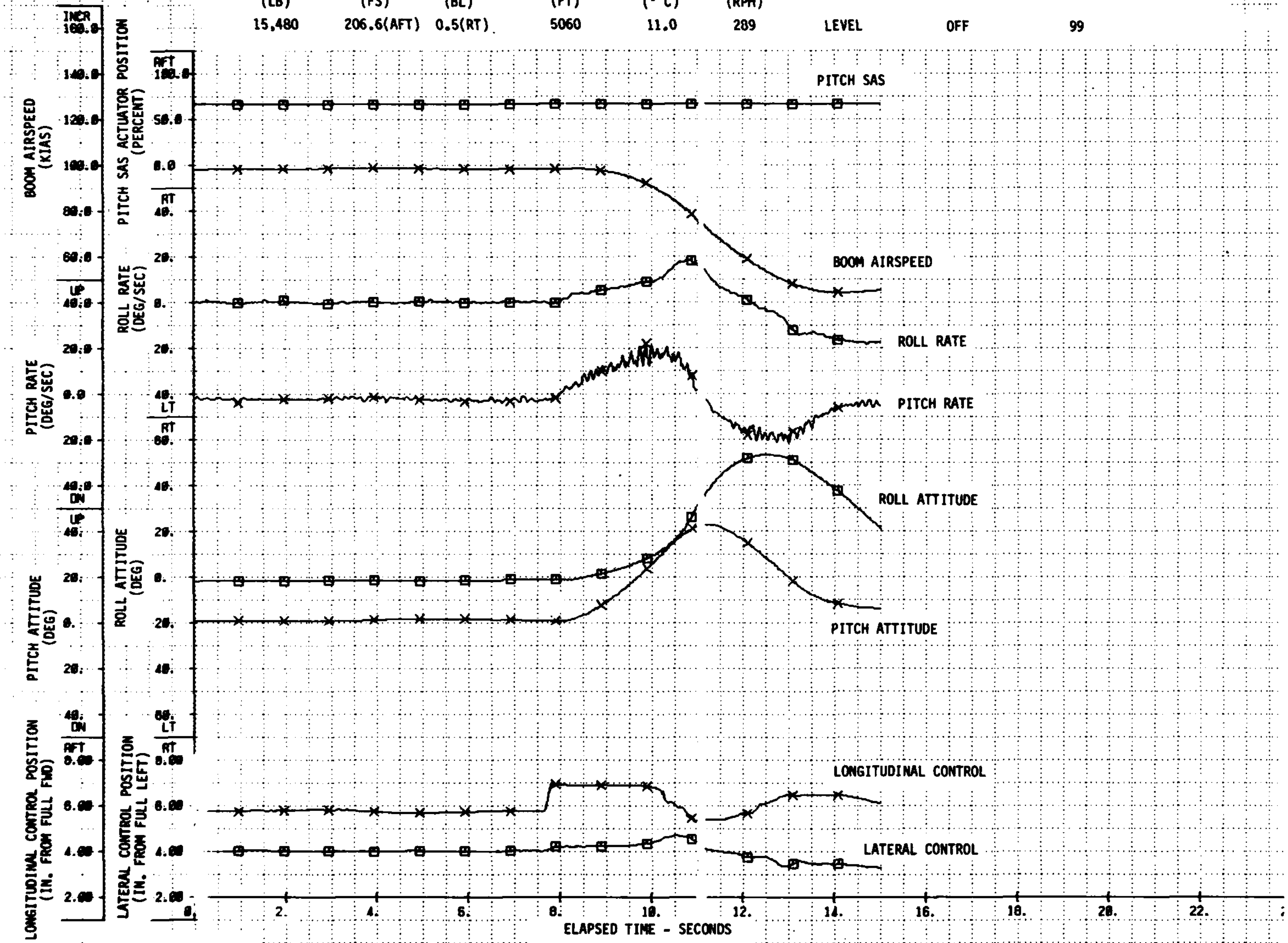


FIGURE 21
LEFT LATERAL STEP
YAH-64 USA S/N 77-23258

AVG. GROSS WEIGHT (LB)	AVG. C.G. LOCATION LONG (FS)	AVG. C.G. LOCATION LAT. (BL)	AVG. DENSITY ALTITUDE (FT)	AVG. OAT (°C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION	TRIM CALIBRATED AIRSPEED (KT)
14,580	206.6 (AFT)	0.5 (RT)	5100	10.5	289	LEVEL	ON	98

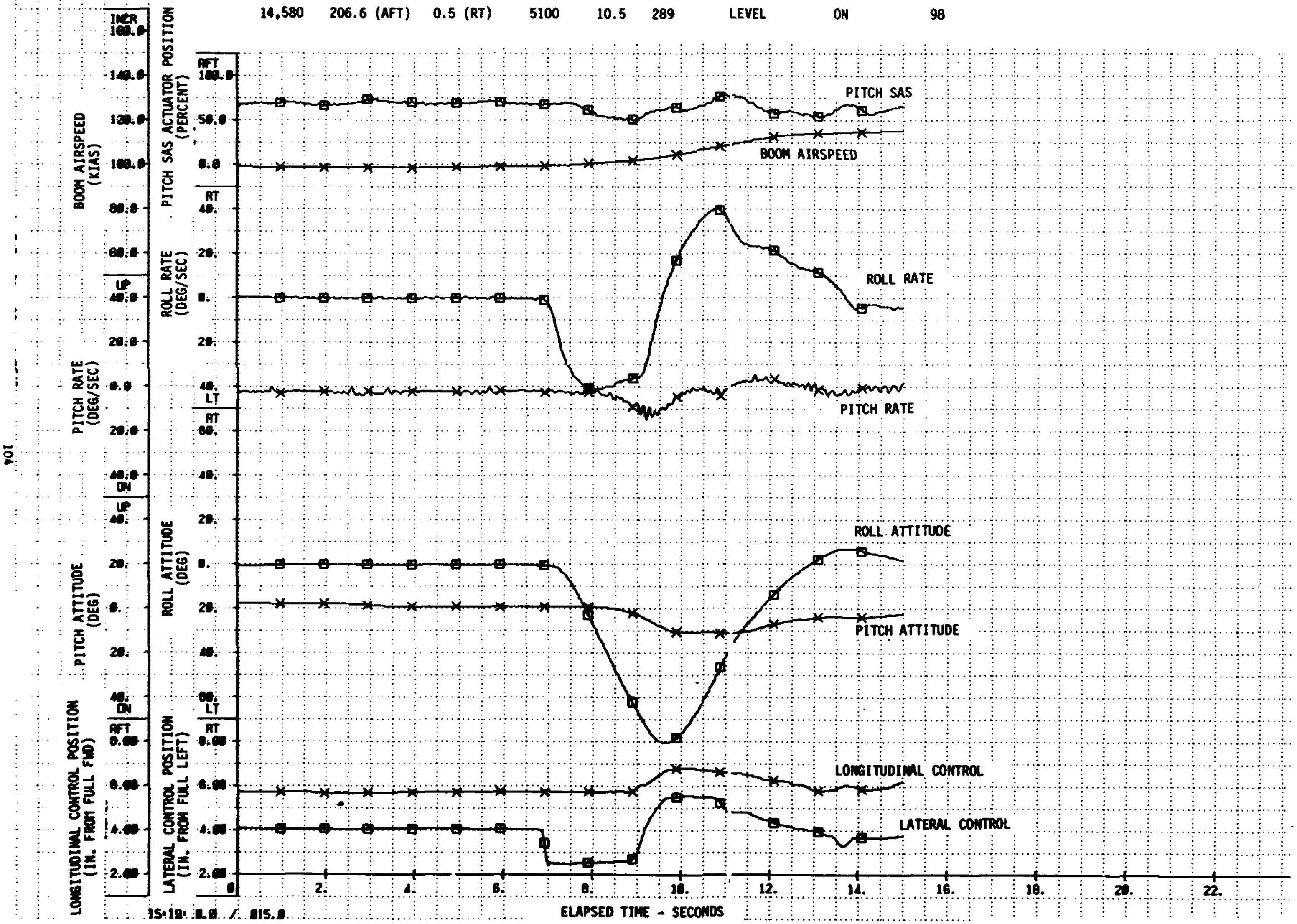
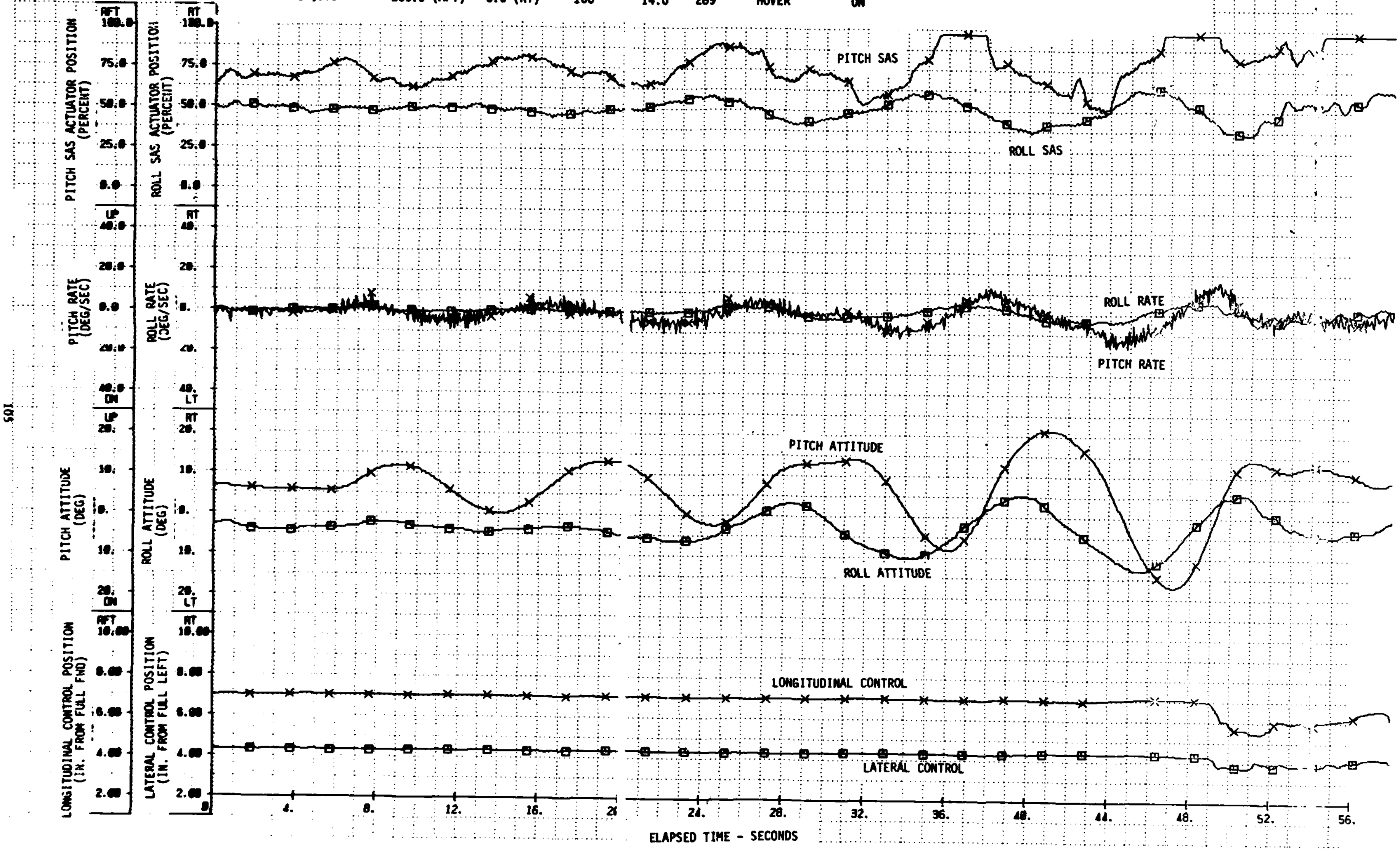


FIGURE 22
AIRCRAFT RESPONSE TO HAS ENGAGEMENT
YAH-64 USA S/N 77-23258

AVG. GROSS WEIGHT (LB)	AVG. C.G. LOCATION LONG (FS)	AVG. C.G. LOCATION LAT. (BL)	AVG. DENSITY ALTITUDE (FT)	AVG. OAT (C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
14,260	206.5 (AFT)	0.5 (RT)	160	14.0	289	HOVER	ON



FIGURL 23
SIMULATED 3 AXIS DASE FAILURE
YAH-64 USA S/N 77-23258

AVG. GROSS WEIGHT (LB)	AVG. C.G. LOCATION LONG (FS)	LAT. (BL)	AVG. DENSITY ALTITUDE (FT)	AVG. OAT (C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
14,260	206.5 (AFT)	0.5 (RT)	180	13.5	289	HOVER	ON

NOTES: 1. AIRCRAFT CONFIGURATION: 8 HELLFIRE
2. HOVER AUGMENTATION SYSTEM: ON

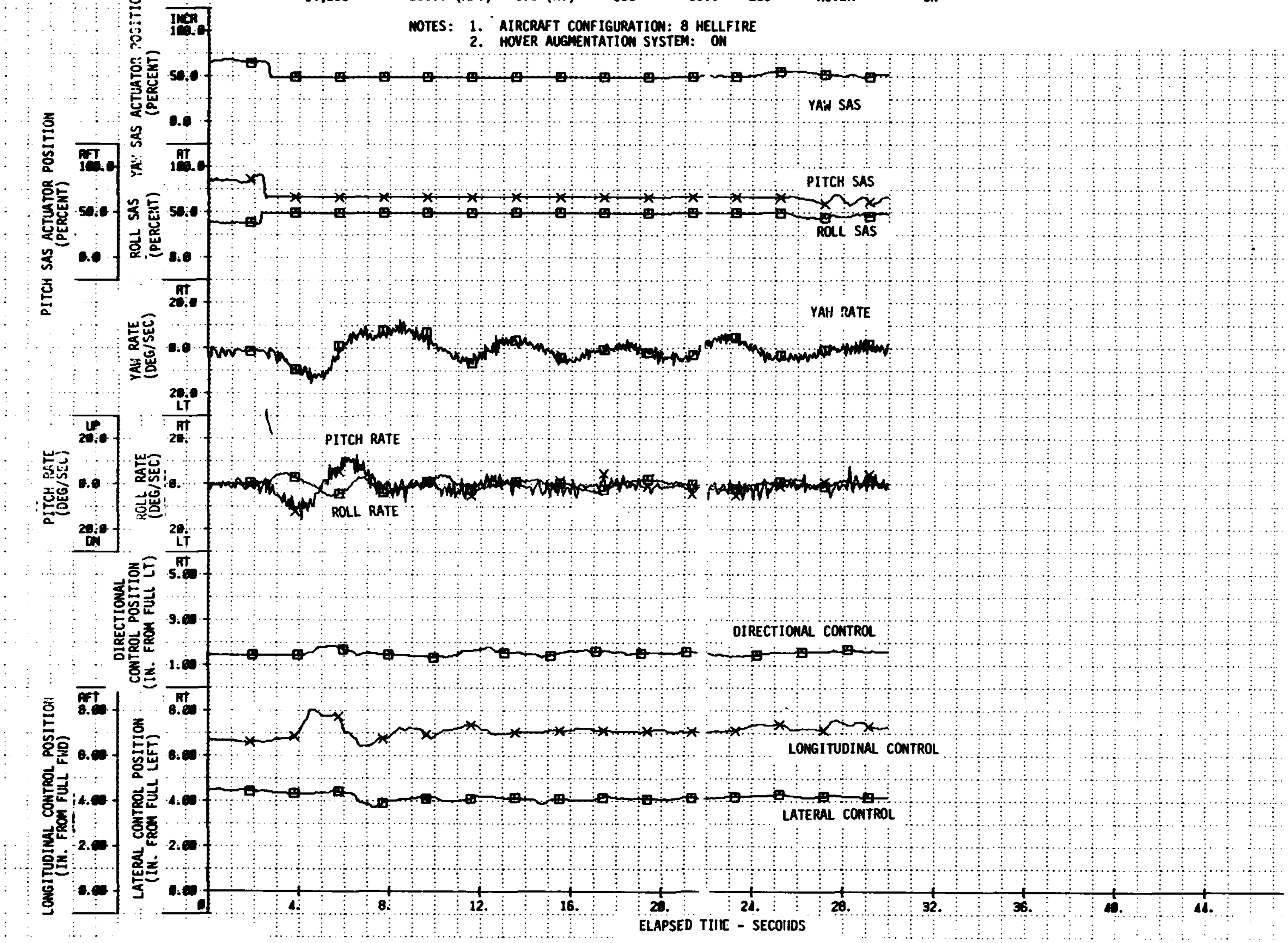


FIGURE 24
STABILIZED REARWARD FLIGHT AT 30 KTAS
YAH-64 USA S/N 77-23258

AVG. GROSS WEIGHT (LB)	AVG. C.G. LOCATION LONG (FS)	AVG. C.G. LOCATION LAT. (BL)	AVG. DENSITY ALTITUDE (FT)	AVG. OAT (C)	TRIM ROTOR SPEED (RPM)	MAIN WHEEL HEIGHT (FT)	DASE CONDITION
14,260	206.5 (AFT)	0.5 (RT)	18?	13.5	289	25	ON

NOTES: 1. AIRCRAFT CONFIGURATION: 8 HELLFIRE
2. HOVER AUGMENTATION SYSTEM: OFF

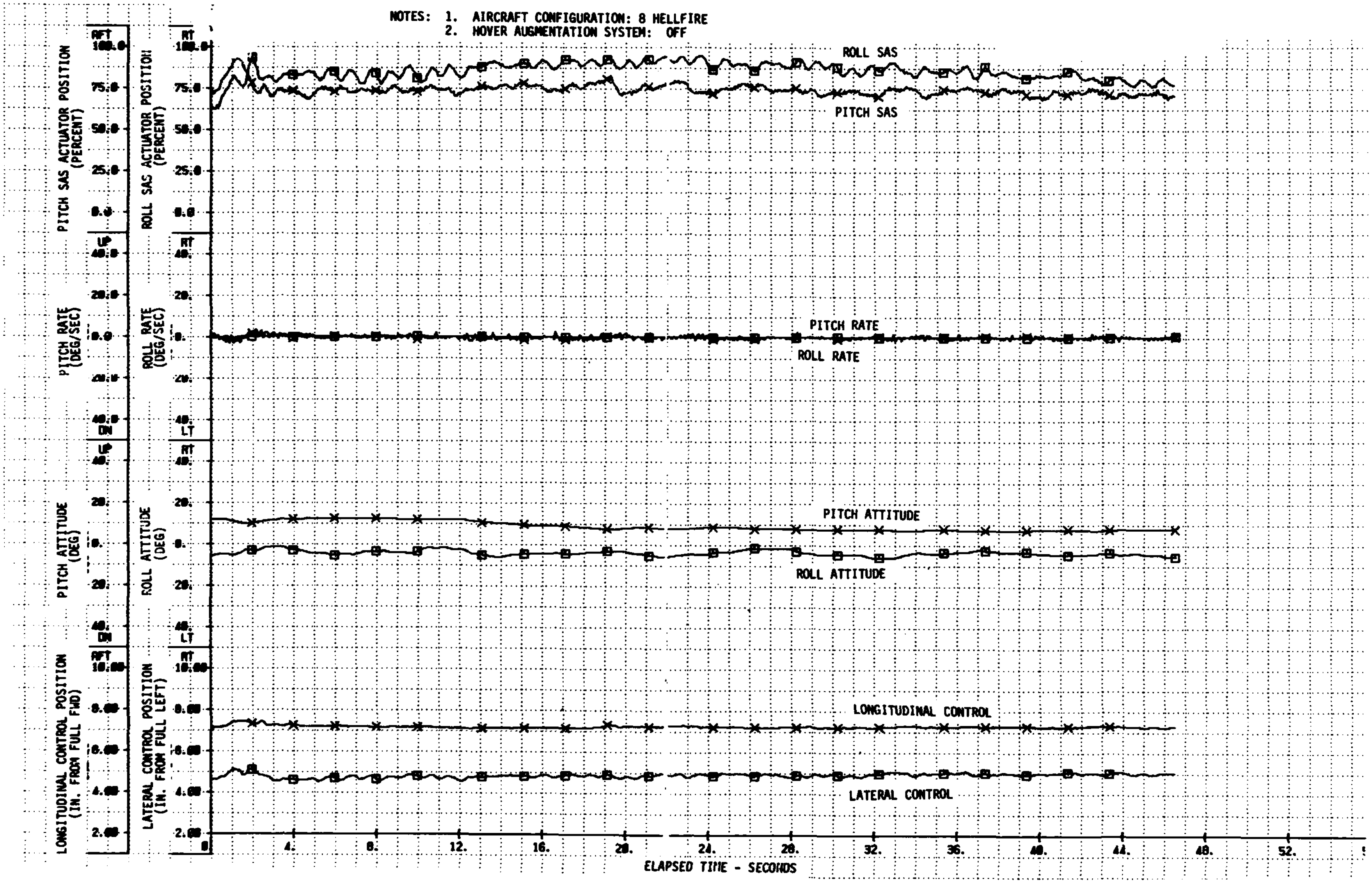


FIGURE 25
AIRCRAFT TRIMMED IN STABILIZED HOVER
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (° C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
14,300	206.5(AFT)	0.5(RT)	180	13.5	289	HOVER	ON

NOTES: 1. AIRCRAFT CONFIGURATION: 8-HELLFIRE
2. HOVER AUGMENTATION SYSTEM: OFF

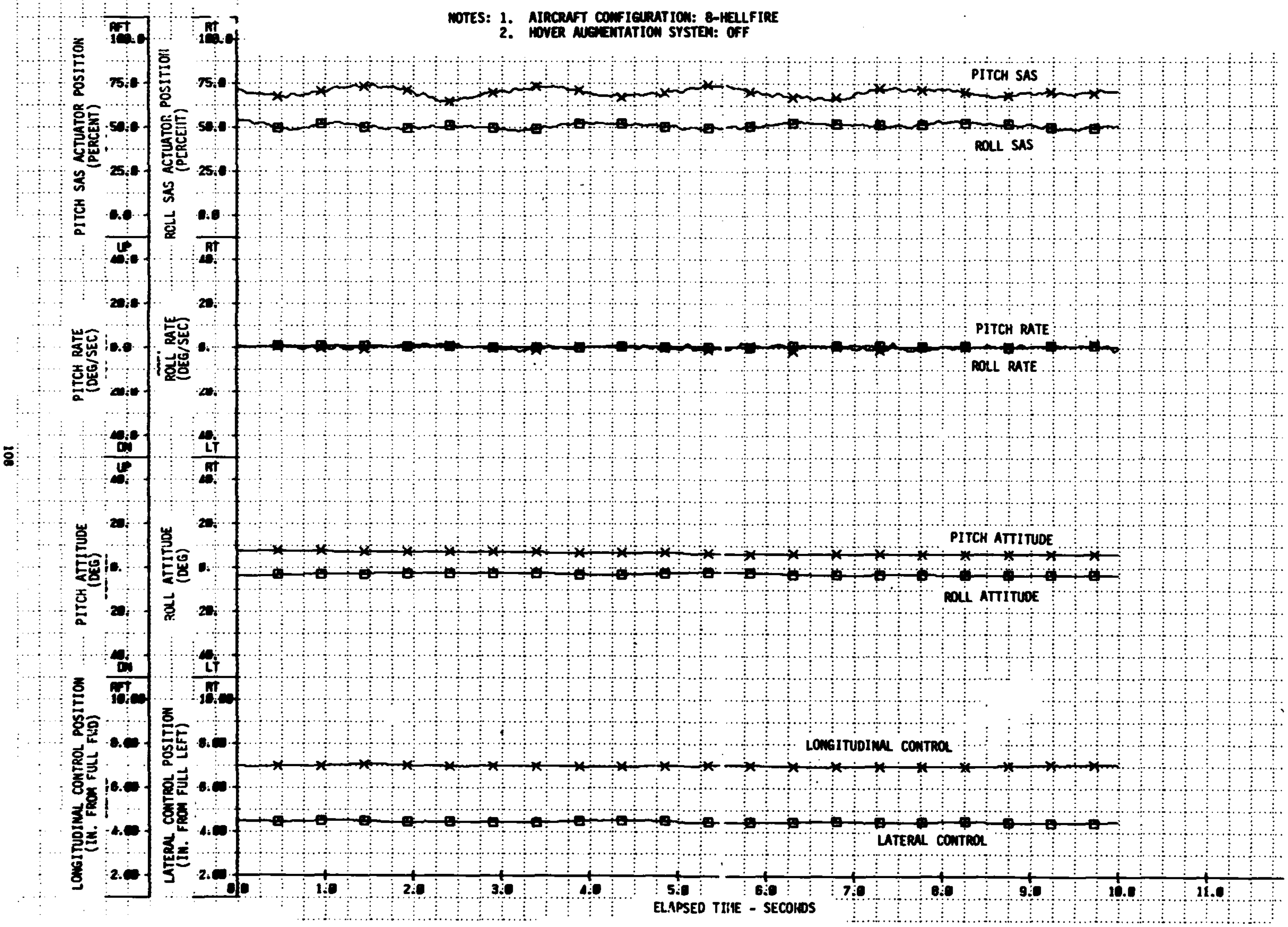
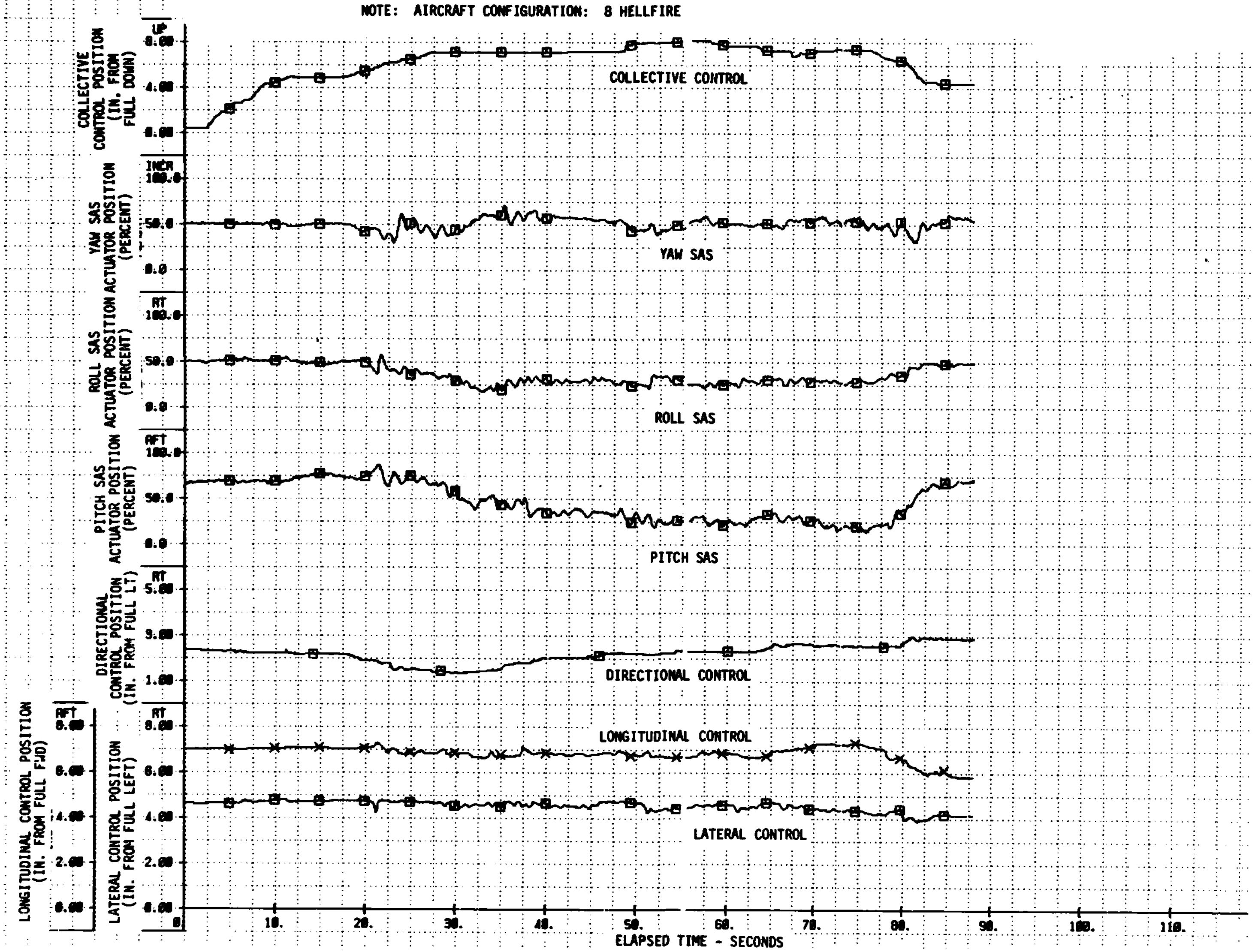
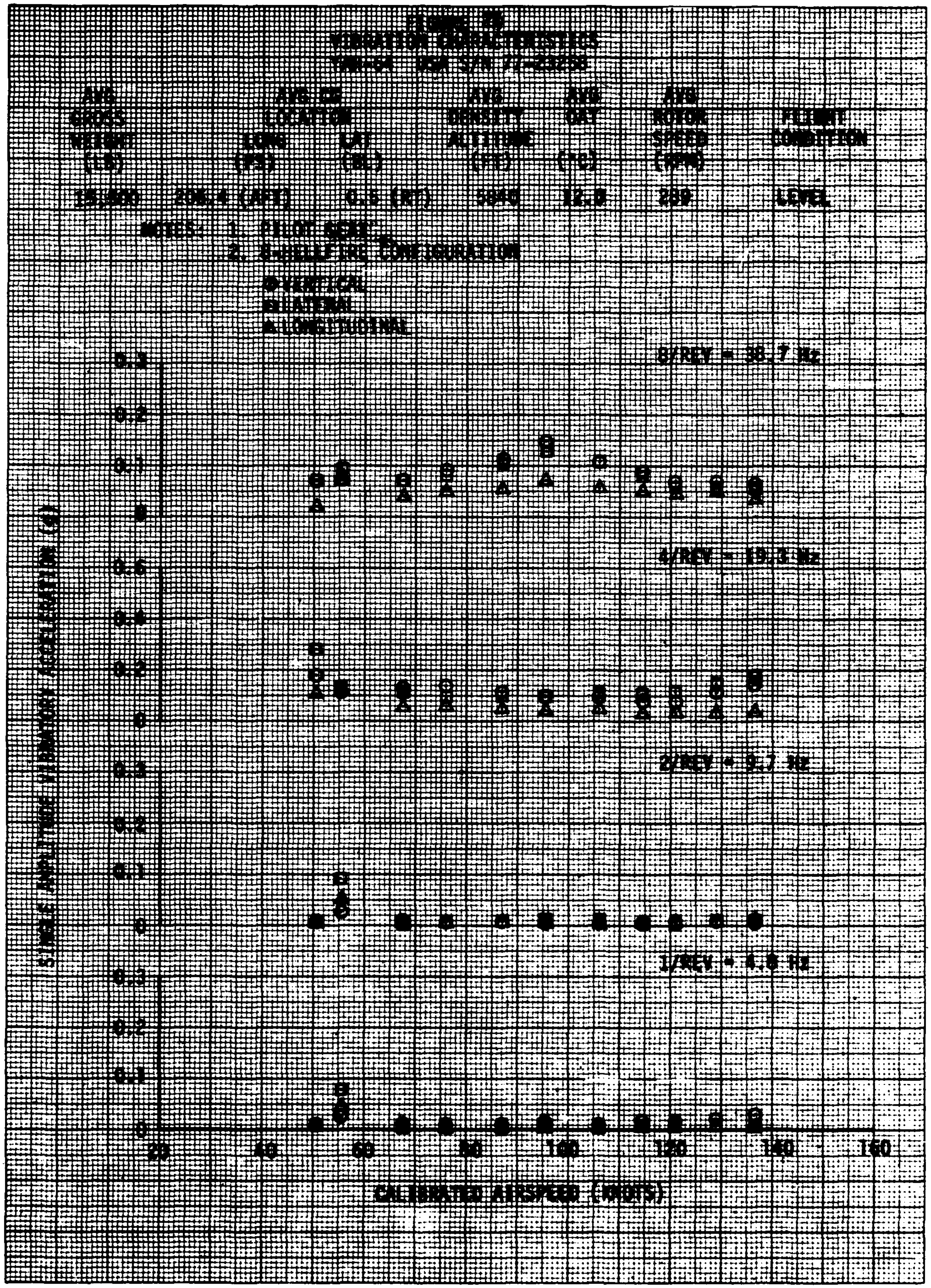


FIGURE 26
INSTRUMENT TAKEOFF
YAH-64 USA S/N 77-23258

AVG. GROSS WEIGHT (LB)	AVG. C.G. LOCATION LONG (FS)	AVG. C.G. LOCATION LAT. (BL)	AVG. DENSITY ALTITUDE (FT)	AVG. OAT (C)	TRIM ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
15,200	205.9 (AFT)	0.5 (RT)	220	13.0	289	HOVER	ON

NOTE: AIRCRAFT CONFIGURATION: 8 HELLFIRE




VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15,500
206.4 (AFT)
0.5 (RT)
5040
12.8
289
LEVEL
NOTES: 1. PILOT SEAT
2. 8-HELLFIRE CONFIGURATION
VERTICAL
LATERAL
LONGITUDINAL
8/REV = 38.7 Hz
4/REV = 19.3 Hz
2/REV = 9.7 Hz
1/REV = 4.8 Hz
SINGLE AMPLITUDE VIBRATORY ACCELERATION (g)
CALIBRATED AIRSPEED (KNOTS)

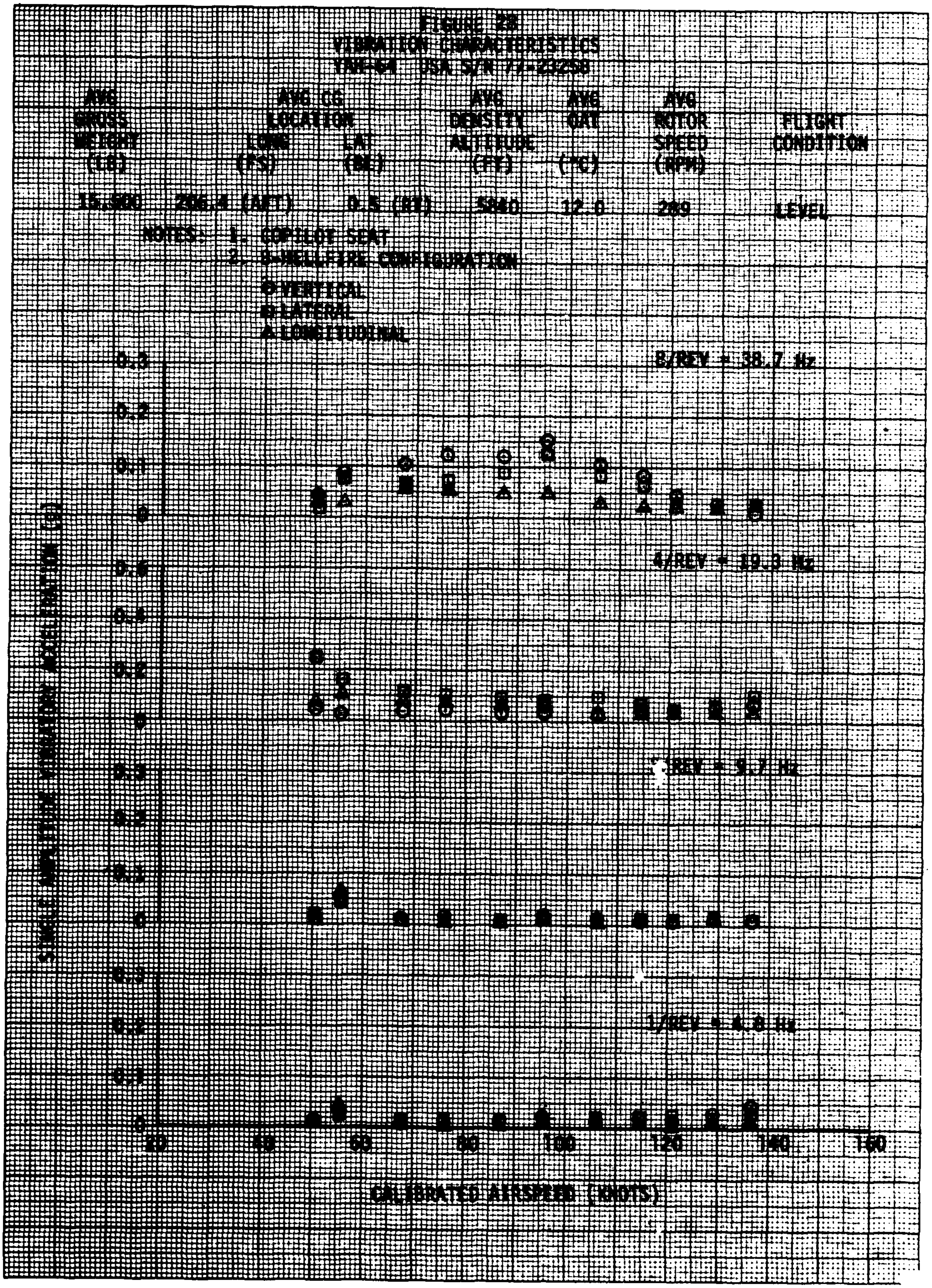


FIGURE 28
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15,500
206.4 (AFT)
0.5 (RT)
5840
12.0
289
LEVEL
NOTES: 1. COPILOT SEAT
2. 8-HELLFIRE CONFIGURATION
VERTICAL
LATERAL
LONGITUDINAL
8/REV = 38.7 Hz
4/REV = 19.3 Hz
REV = 9.7 Hz
1/REV = 4.8 Hz
0.3
0.2
0.1
0
SINGLE AMPLITUDE VIBRATION ACCELERATION (G)
20
40
60
80
100
120
140
160
CALIBRATED AIRSPEED (KNOTS)

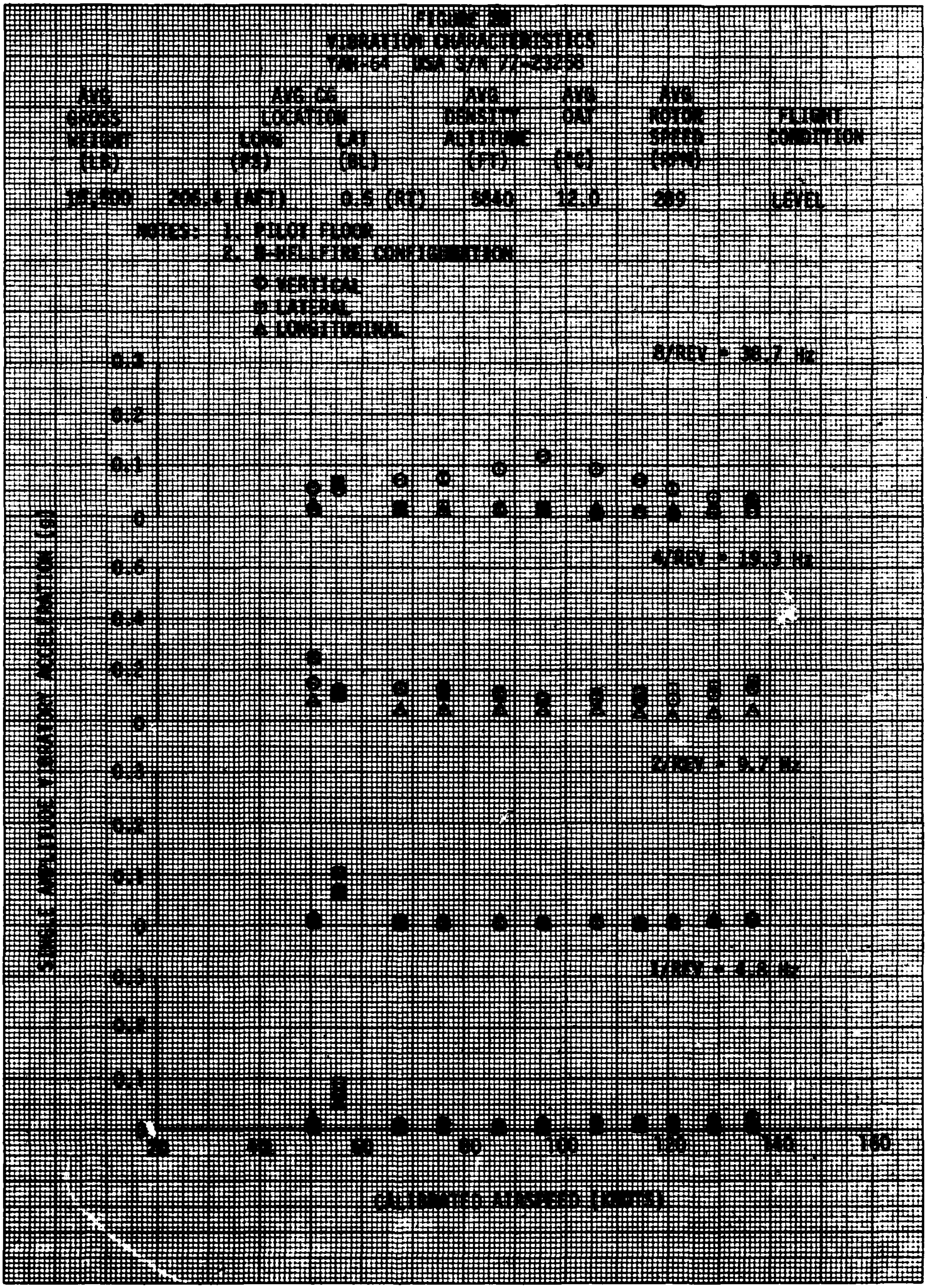
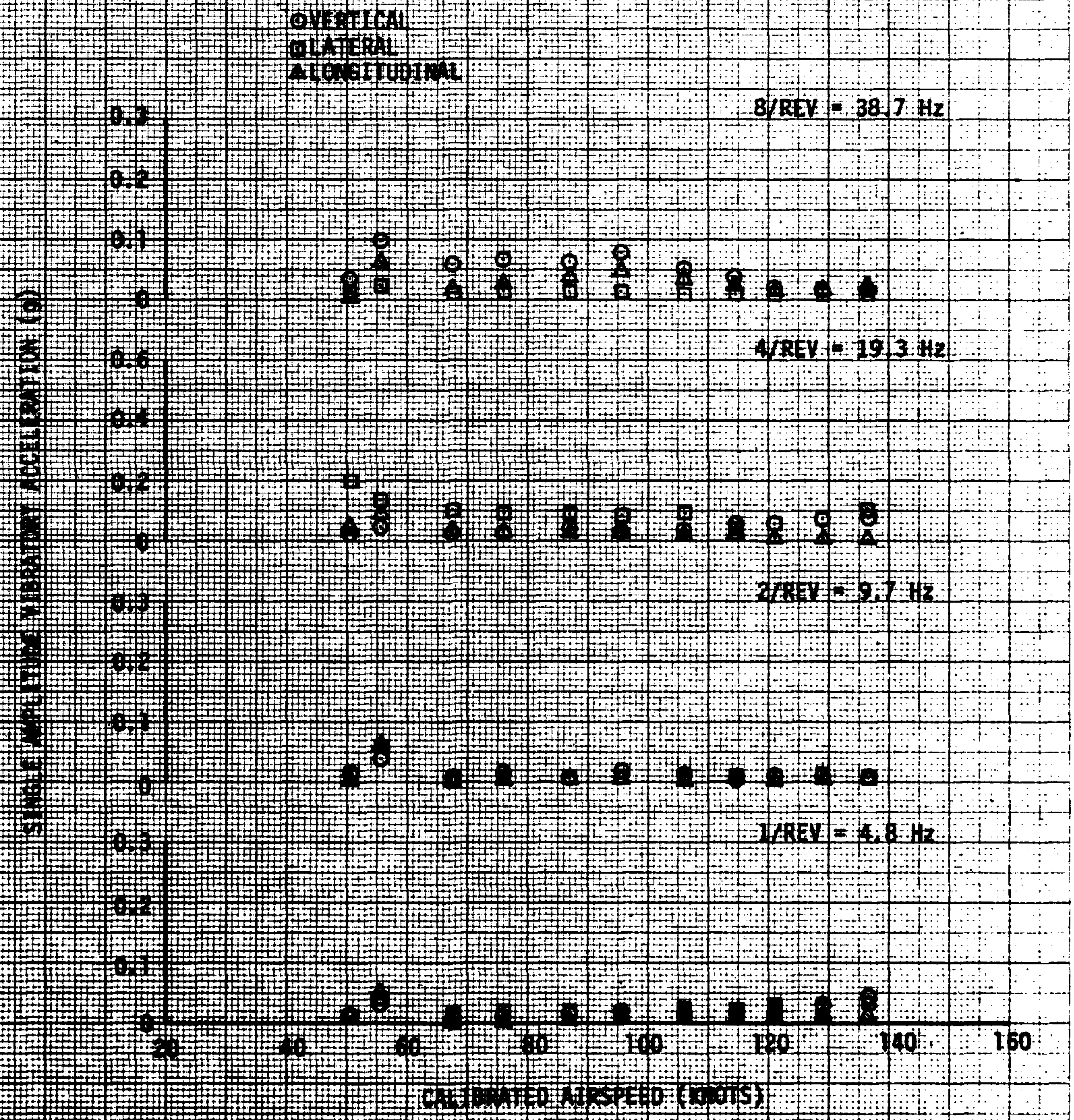

VIBRATION CHARACTERISTICS
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
0.5 (RT)
5840
12.0
LEVEL
NOTES: 1. PILOT FLOOR
2. HELLFIRE CONFIGURATION
VERTICAL
LATERAL
LONGITUDINAL
8/REV = 38.7 Hz
4/REV = 19.3 Hz
2/REV = 9.7 Hz
1/REV = 4.8 Hz
SINGLE AMPLITUDE VIBRATORY ACCELERATION (G)
CALIBRATED AIRSPEED (KNOTS)
20
40
60
80
100
120
140
160

FIGURE 30
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,500	206.4 (AFT)	0.5 (RT)	5840	12.0	289	LEVEL

NOTES: 1. COPILOT FLOOR
2. 8-HELLFIRE CONFIGURATION



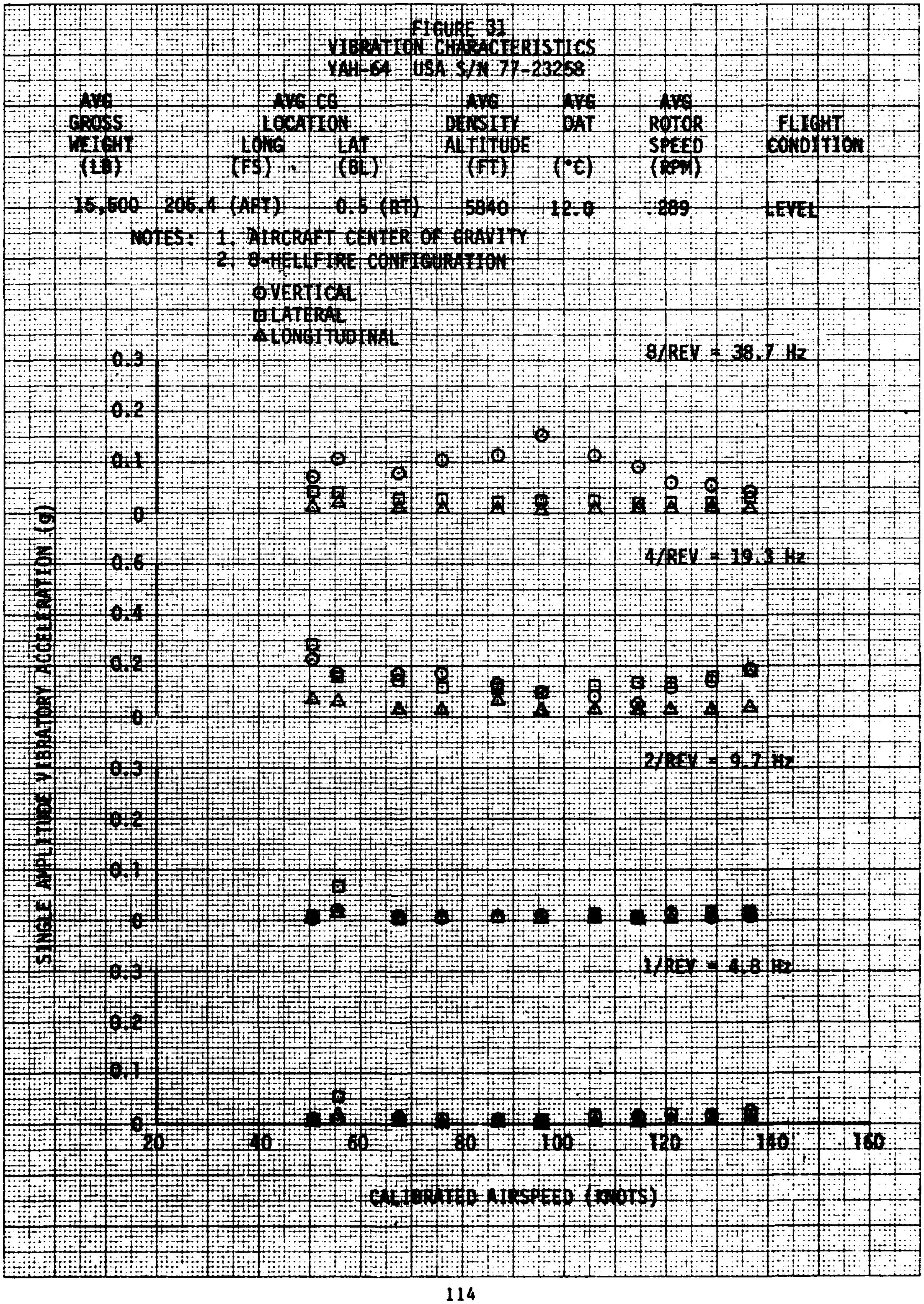
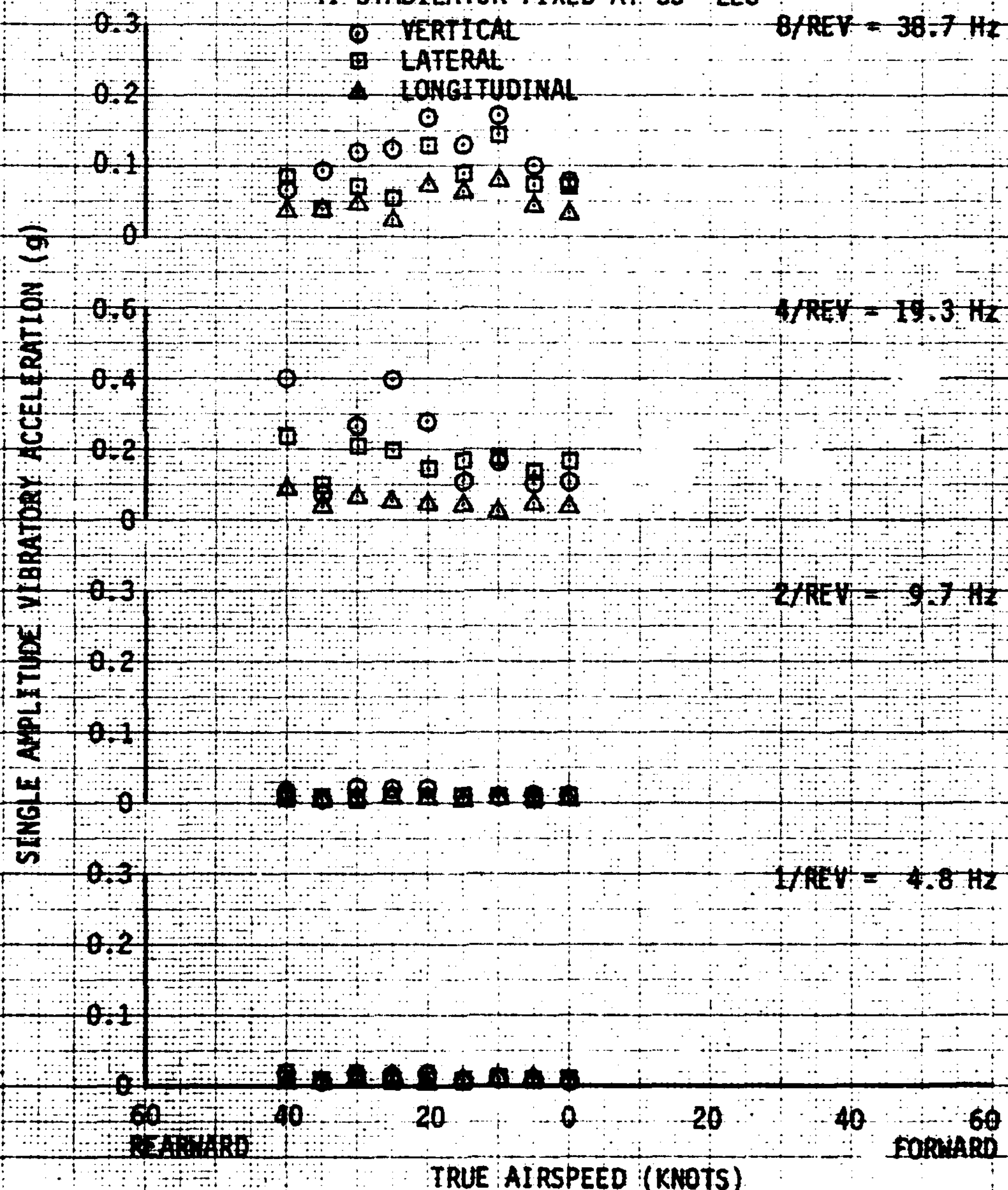


FIGURE 31
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15,500
206.4 (AFT)
0.5 (RT)
5840
12.0
289
LEVEL
NOTES: 1. AIRCRAFT CENTER OF GRAVITY
2. 8-HELLFIRE CONFIGURATION
VERTICAL
LATERAL
LONGITUDINAL
8/REV = 38.7 Hz
4/REV = 19.3 Hz
2/REV = 9.7 Hz
1/REV = 4.8 Hz
SINGLE AMPLITUDE VIBRATORY ACCELERATION (g)
CALIBRATED AIRSPEED (KNOTS)
0
0.1
0.2
0.3
0.4
0.6
20
40
60
80
100
120
140
160

FIGURE 32
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,020	206.5 (AFT)	0.5 (RT)	160	13.5	289	REARWARD

NOTES: 1. PILOT SEAT
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 35° LEU



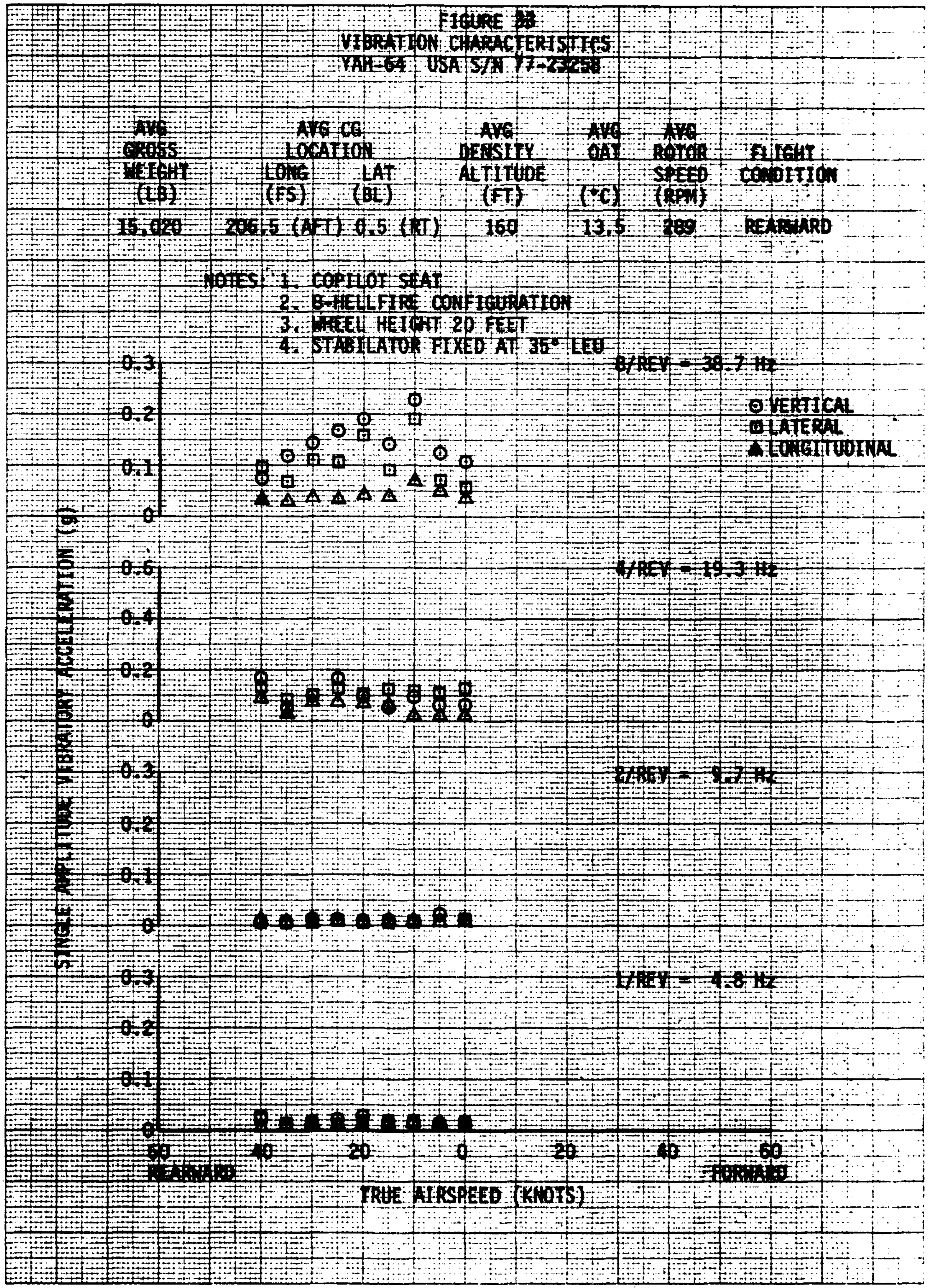


FIGURE 38
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15,020
206.5 (AFT)
0.5 (RT)
160
13.5
289
REARWARD
NOTES: 1. COPILOT SEAT
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 35° LED
8/REV = 38.7 Hz
4/REV = 19.3 Hz
2/REV = 9.7 Hz
1/REV = 4.8 Hz
VERTICAL
LATERAL
LONGITUDINAL
SINGLE AMPLITUDE VIBRATORY ACCELERATION (g)
0.3
0.2
0.1
0
0.6
0.4
0.2
0
0.3
0.2
0.1
0
0.3
0.2
0.1
0
60
40
20
0
20
40
60
REARWARD
FORWARD
TRUE AIRSPEED (KNOTS)

FIGURE 38
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,020	206.5 (AFT)	0.5 (RT)	160	13.5	289	REARWARD

NOTES: 1. PILOT FLOOR
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 35° LEU

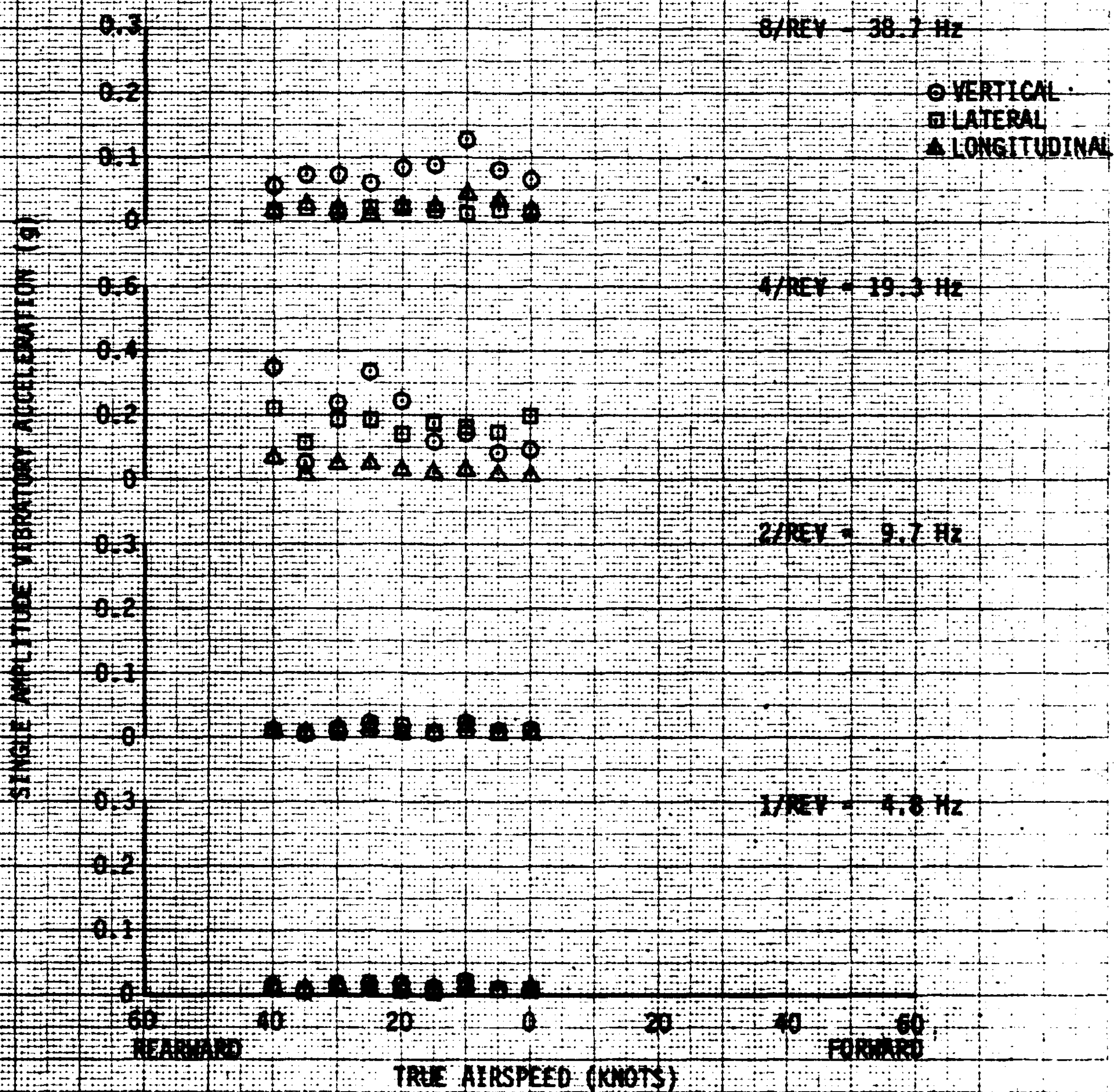


FIGURE 35
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,020	206.5 (AFT)	0.5 (RT)	160	13.5	289	REARWARD

NOTES: 1. COPILOT FLOOR
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 35° LEU

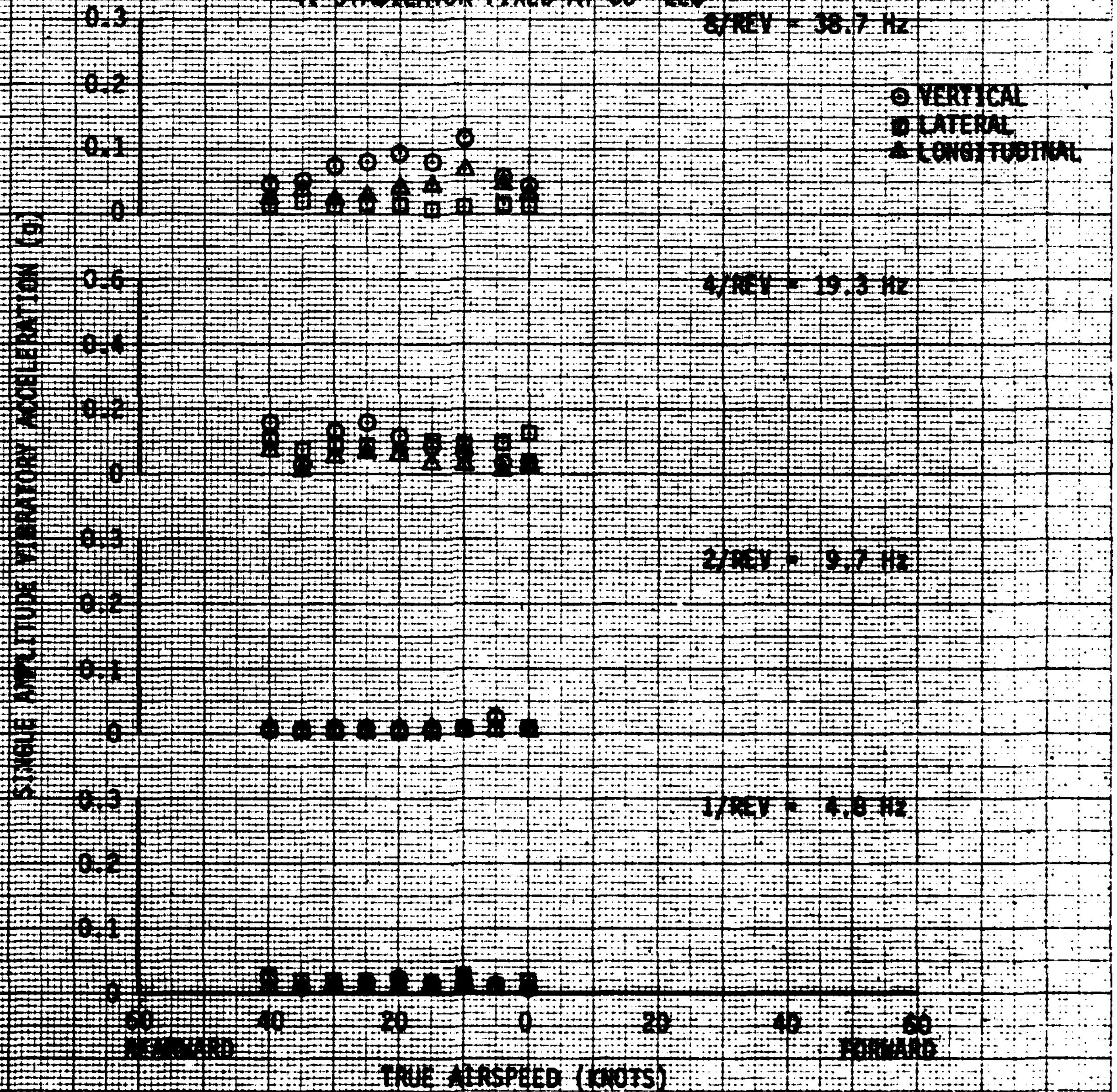


FIGURE 36
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15,020	206.5 (AFT)	0.5 (RT)	160	13.5	289	REARWARD

NOTES: 1. AIRCRAFT CENTER OF GRAVITY
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 35° LEU

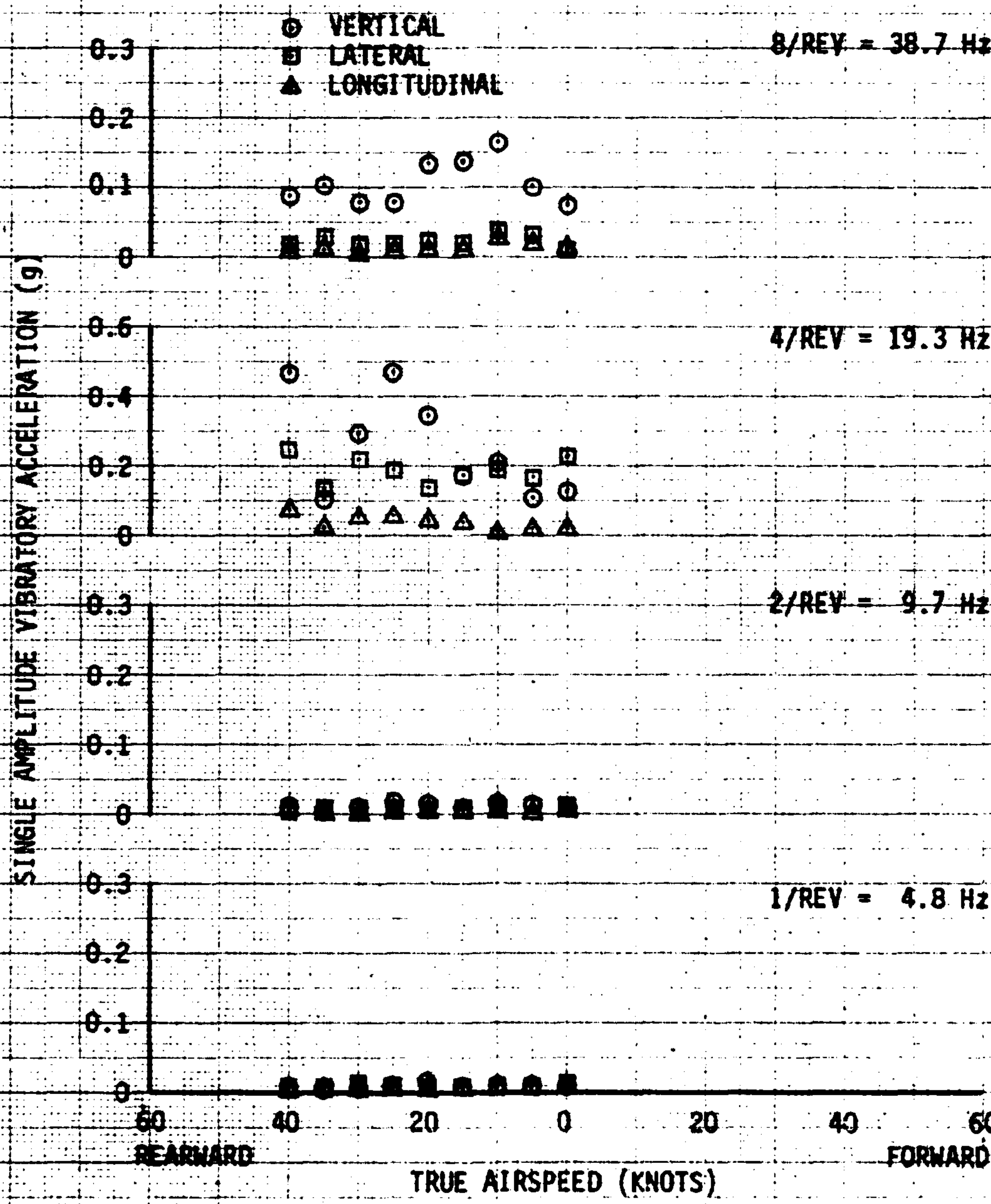


FIGURE 37
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,720	206.5 (AFT)	0.5 (RT)	200	14.0	289	REARWARD

NOTES : 1. PILOT SEAT
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 5° LED

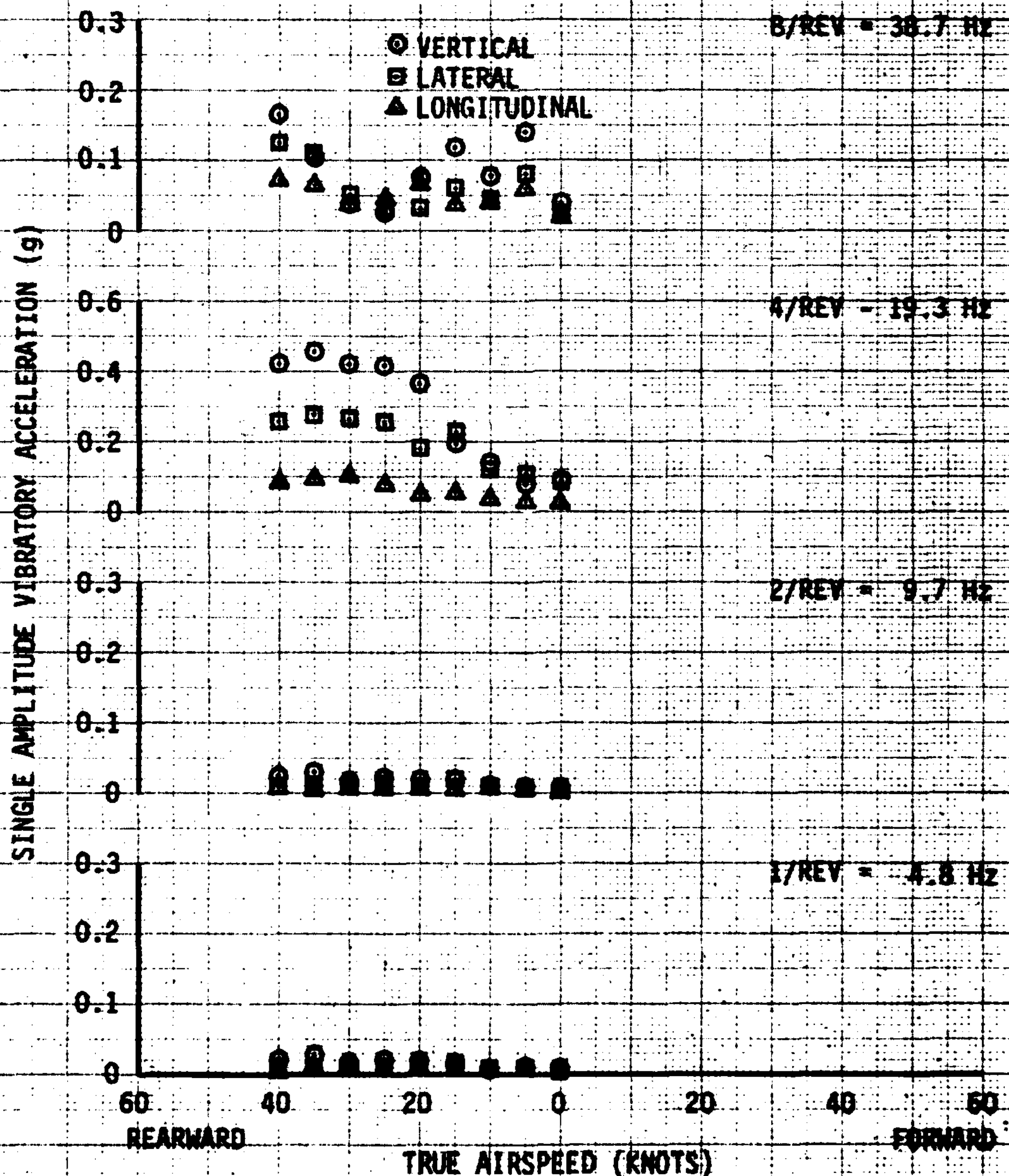


FIGURE 38
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14.720	206.5 (AFT)	0.5 (RT)	200	14.0	289	REARWARD

NOTES: 1. CO/PILOT SEAT
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 5° LED

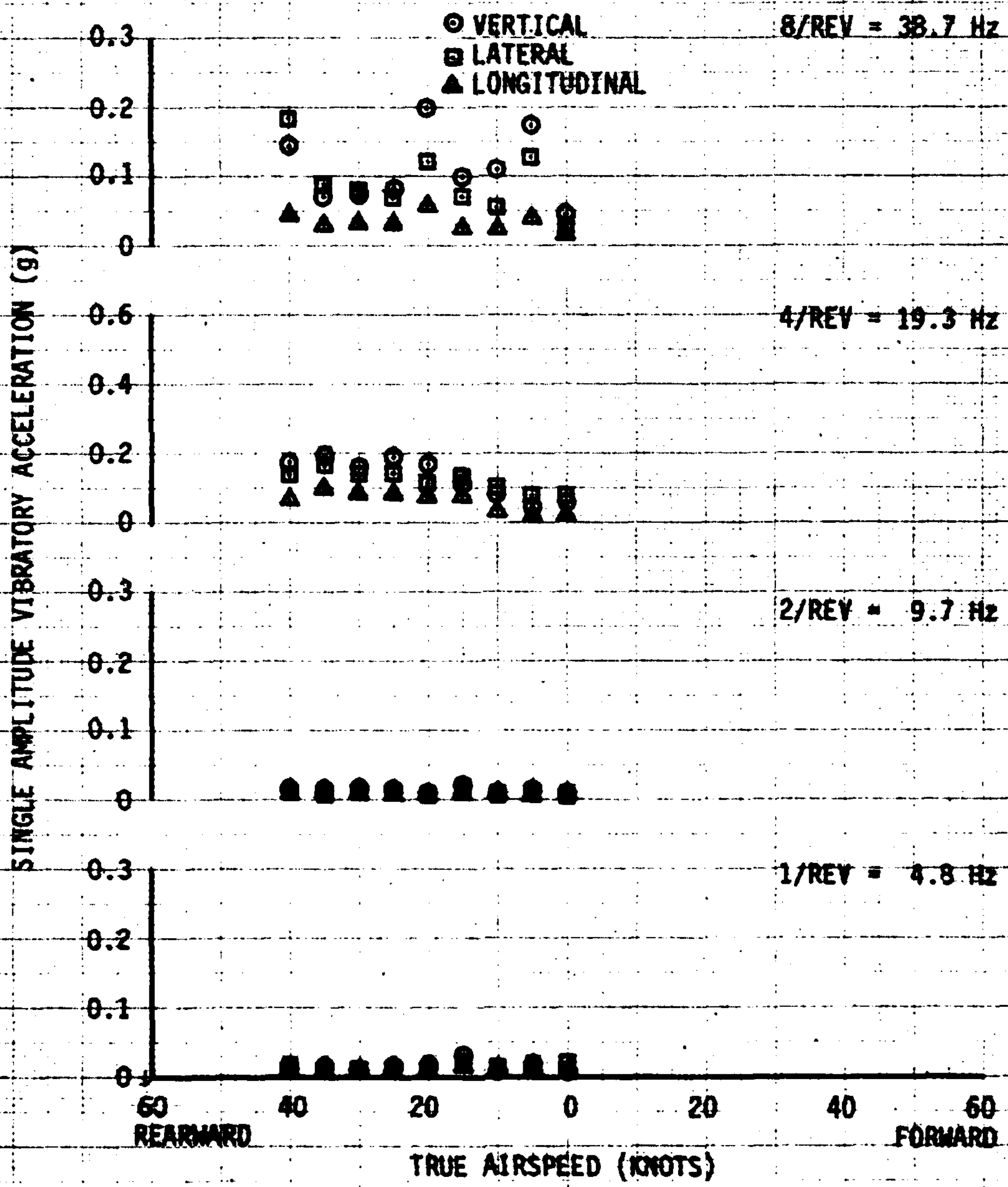


FIGURE 39
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,720	206.5 (AFT)	0.5 (RT)	200	14.0	289	REARWARD

NOTES: 1. PILOT FLOOR
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 5° LED

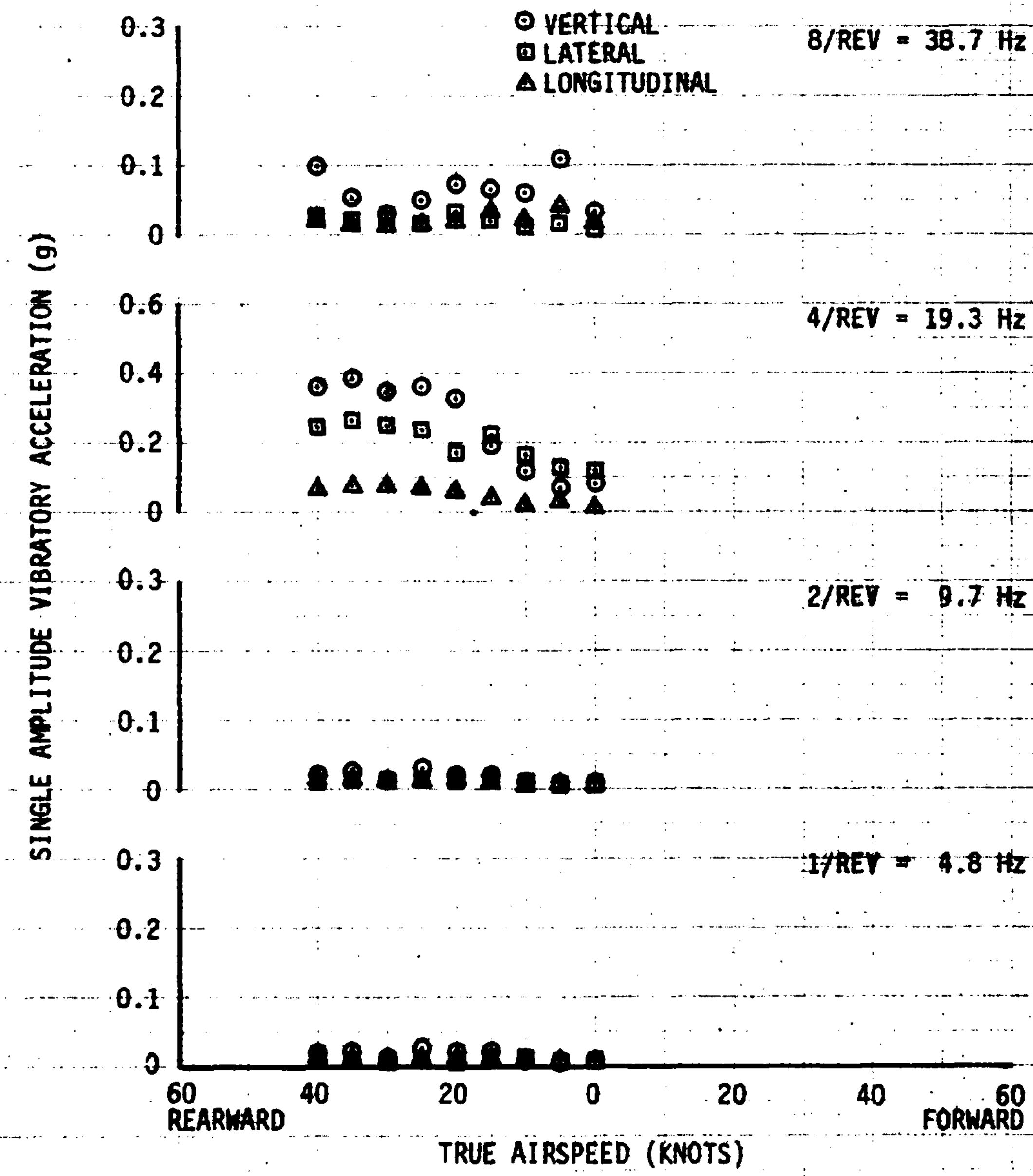


FIGURE 40
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,720	206.5 (AFT)	0.5 (RT)	200	14.0	289	REARWARD

NOTES: 1. CO/PILOT FLOOR
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 5° LED

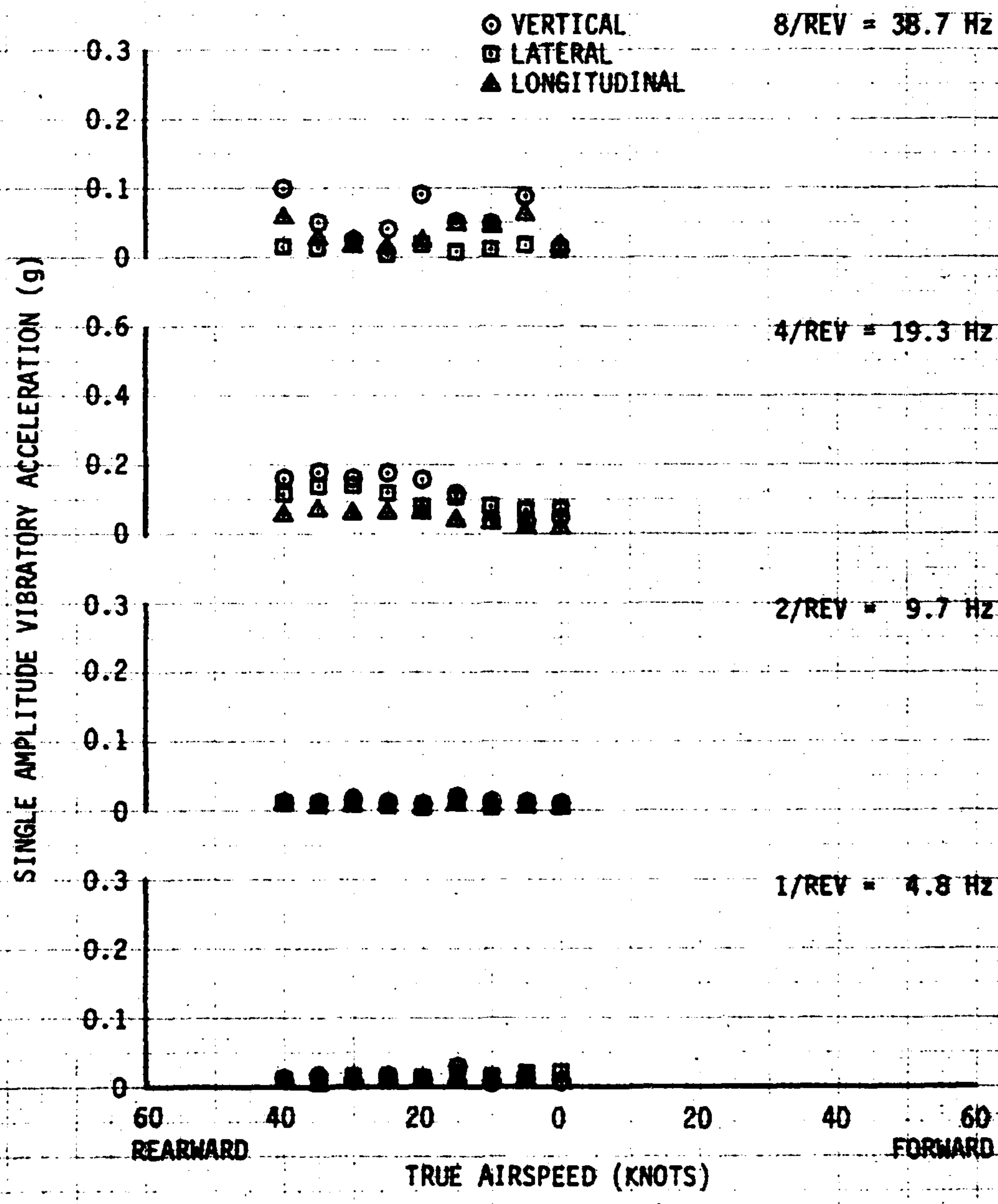


FIGURE 41
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,720	206.5 (AFT)	0.5 (RT)	200	14.0	289	REARWARD

NOTES: 1. AIRCRAFT CENTER OF GRAVITY
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. STABILATOR FIXED AT 5° LED

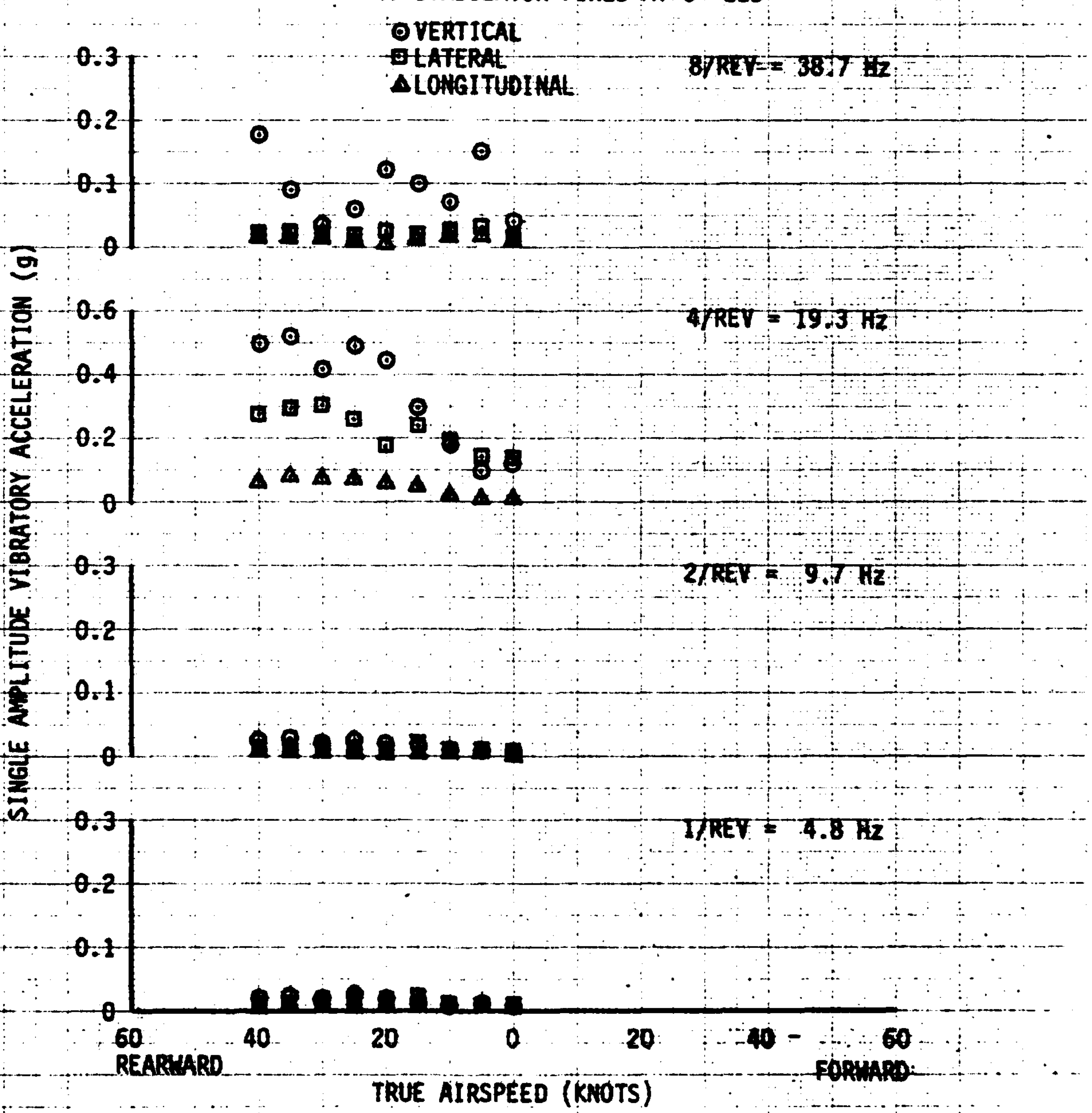


FIGURE 42
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,460	206.5 (AFT)	0.5 (RT)	260	14.5	289	REARWARD

NOTES: 1. PILOT SEAT
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. TRUE AIRSPEED 30 KNOTS

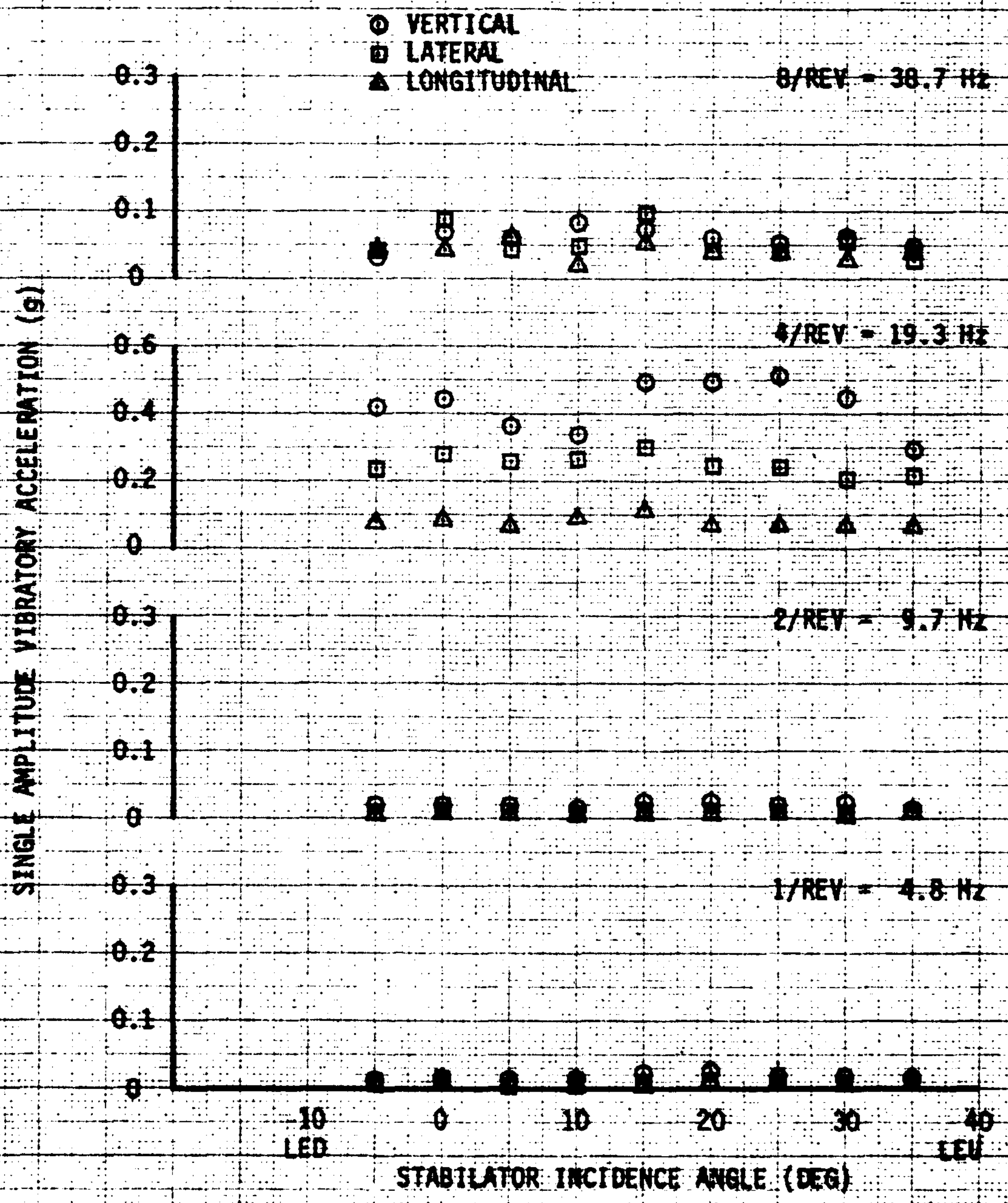


FIGURE 43
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,460	206.5 (AFT)	0.5 (RT)	260	14.5	289	REARWARD

NOTES: 1. COPILOT SEAT
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. TRUE AIRSPEED 30 KNOTS

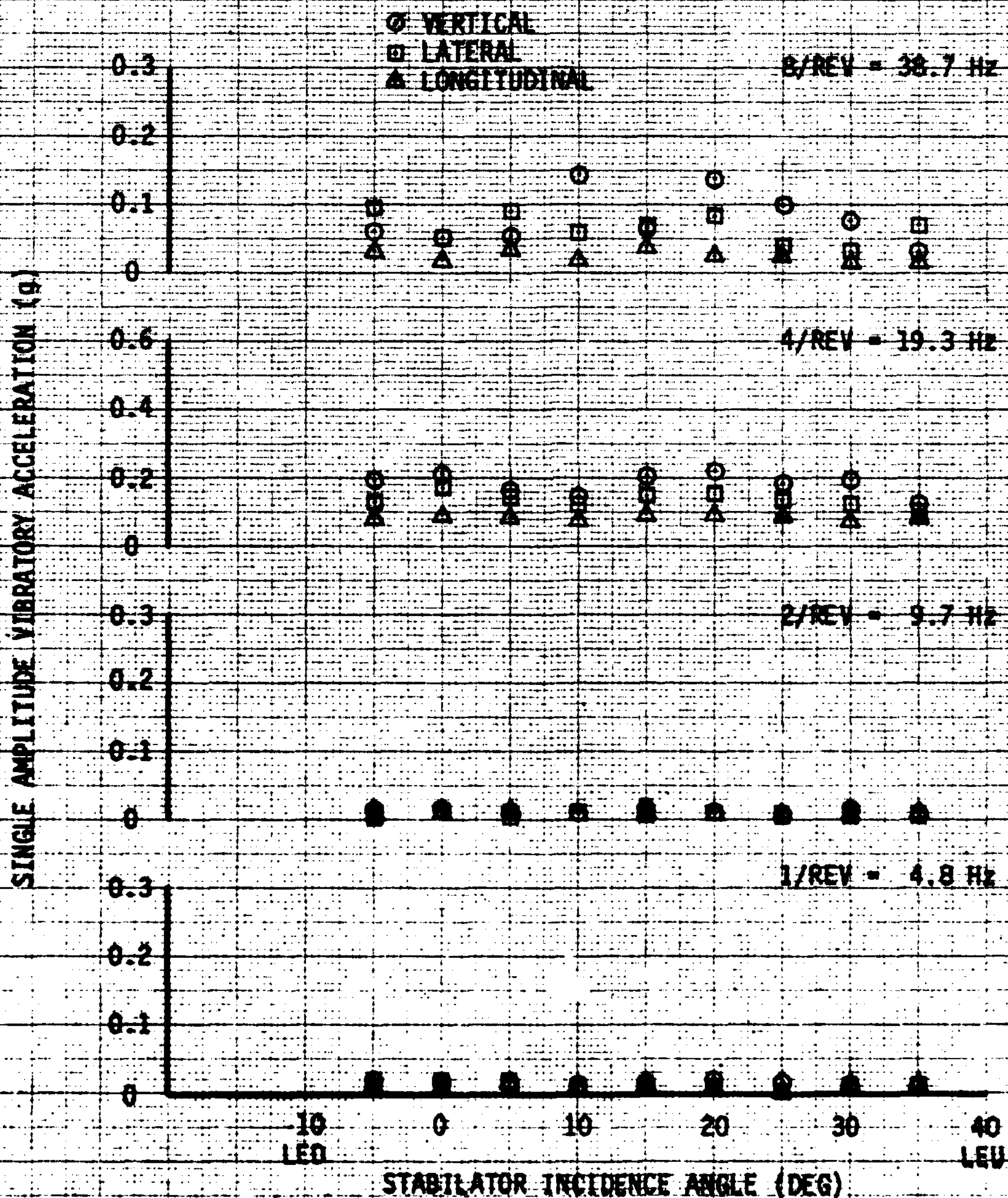


FIGURE 48
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,460	206.5 (AFT)	0.5 (RT)	260	14.5	289	REARWARD

NOTES: 1. PILOT FLOOR
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. TRUE AIRSPEED 30 KNOTS

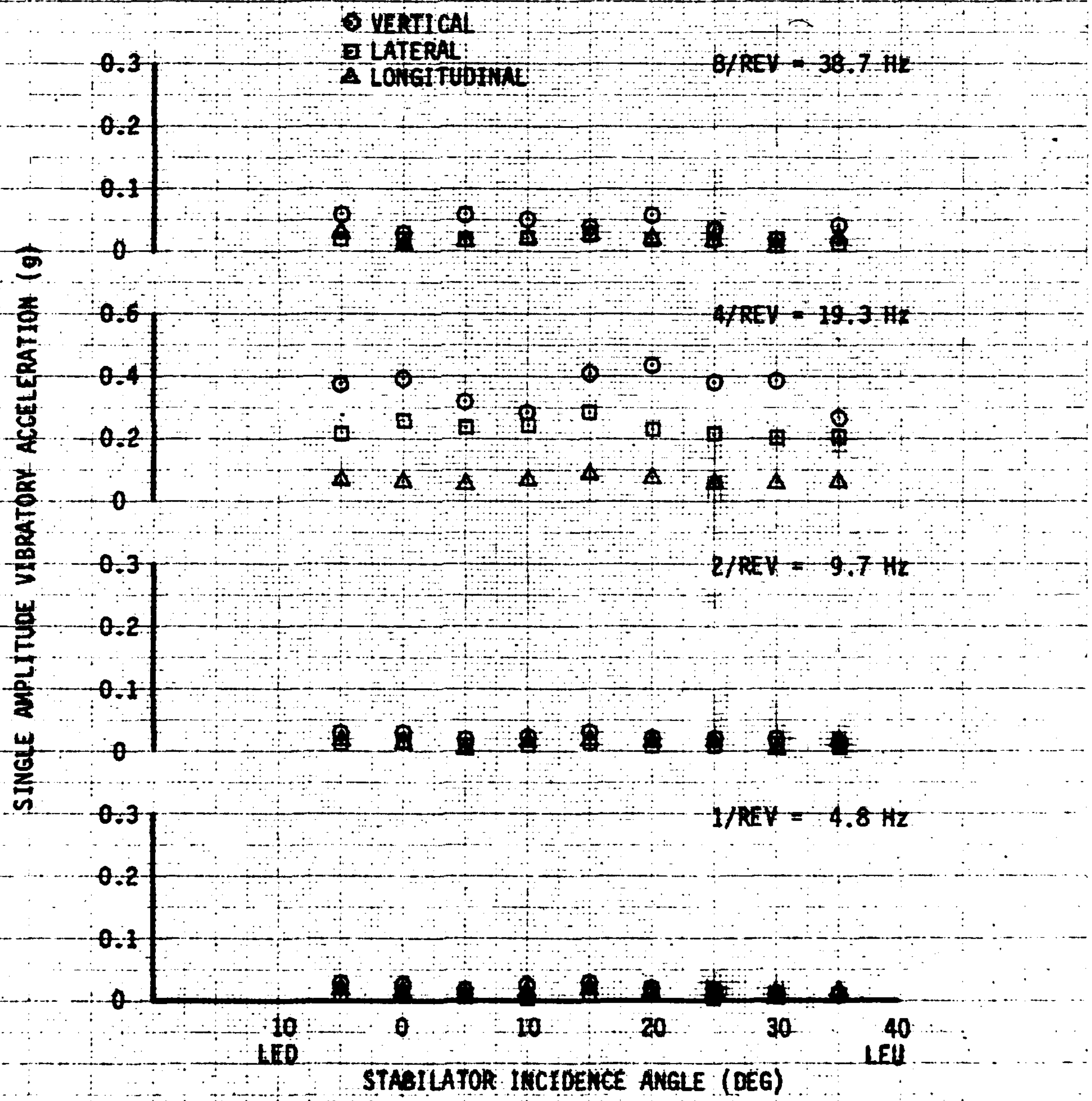


FIGURE 48
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,460	206.5 (AFT)	0.5 (RT)	260	14.5	289	REARWARD

NOTES: 1. COPILOT FLOOR
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. TRUE AIRSPEED 30 KNOTS

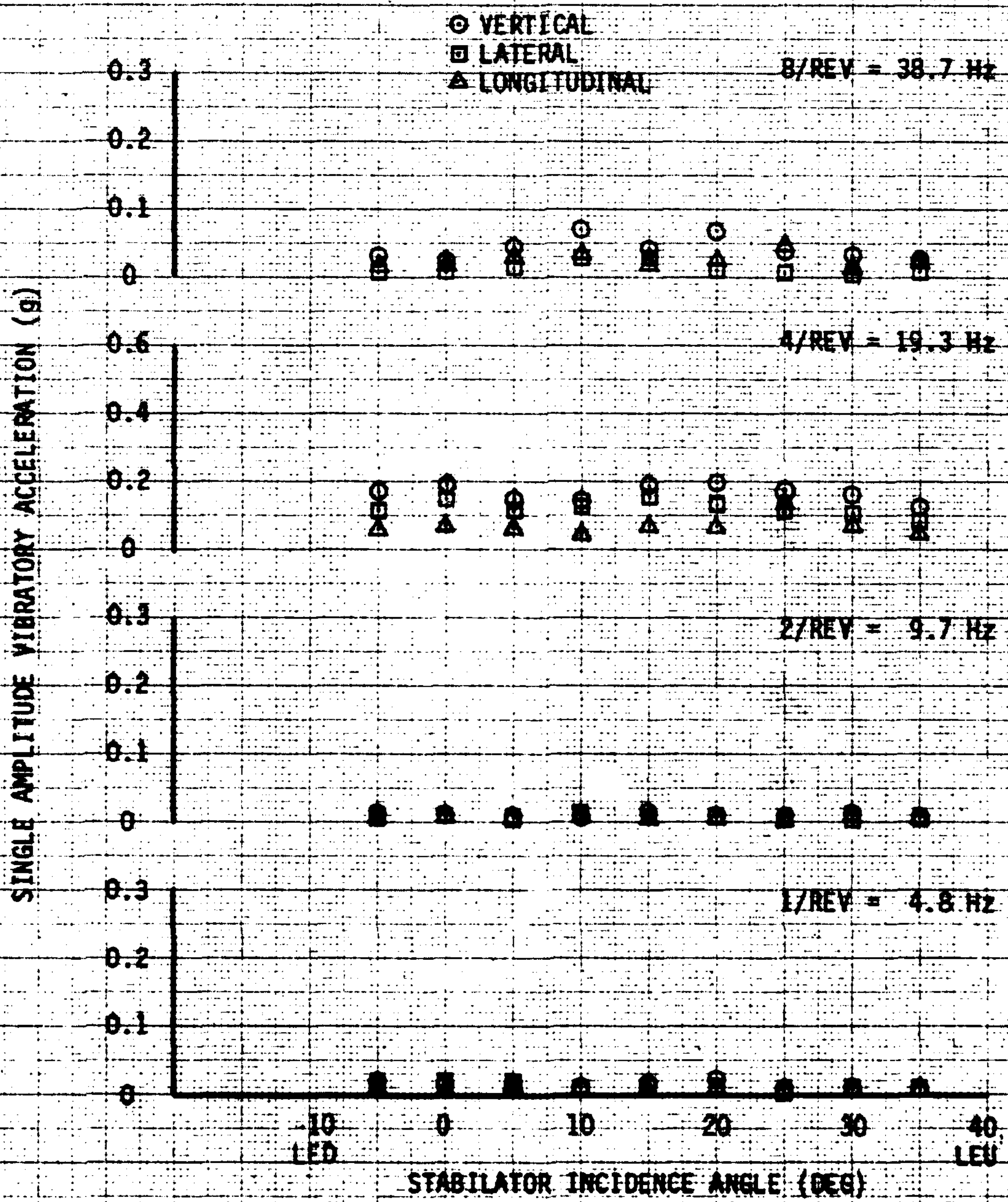
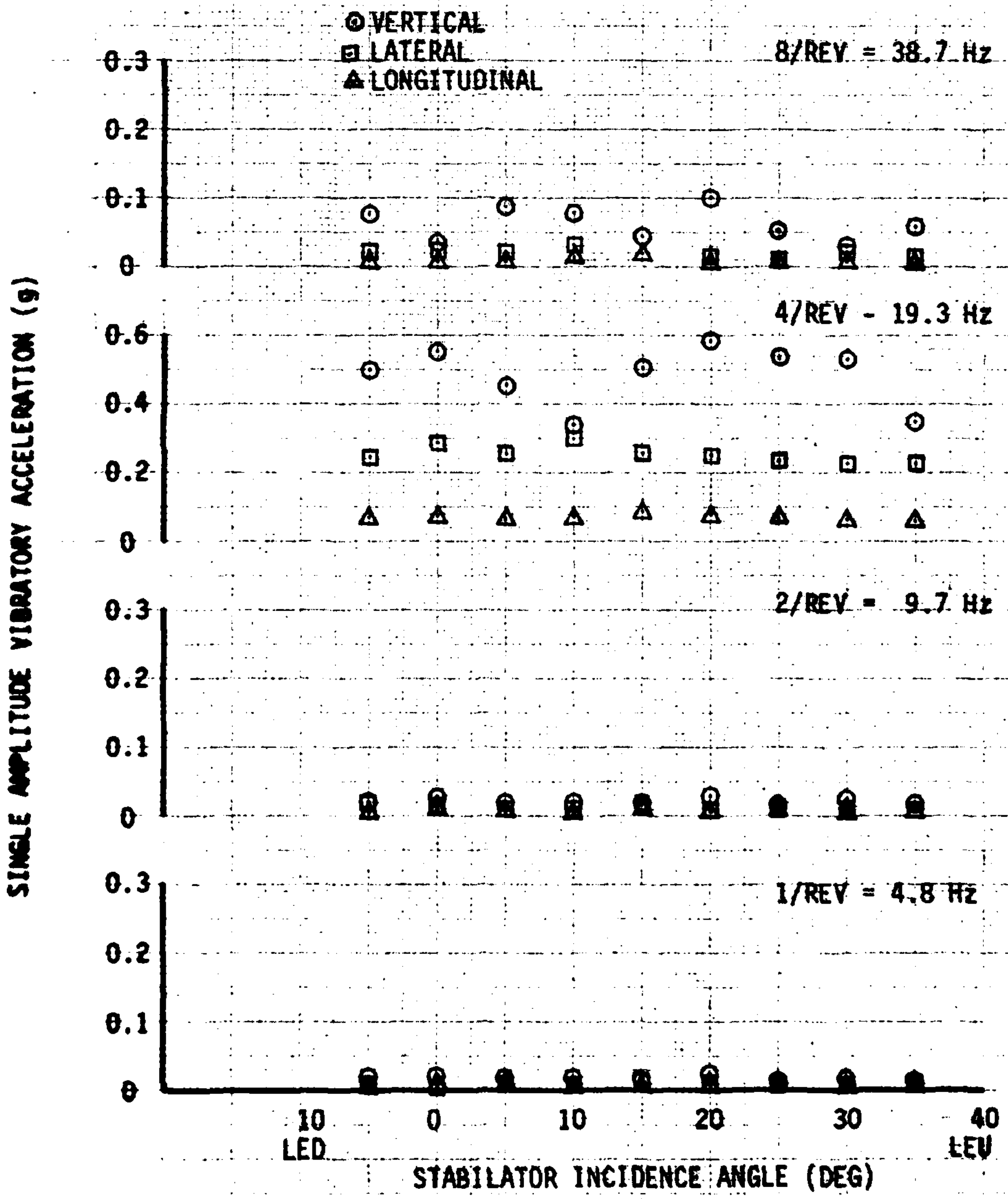


FIGURE 46
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14,460	206.5 (AFT)	0.5 (RT)	260	14.5	289	REARWARD

NOTES: 1. AIRCRAFT CENTER OF GRAVITY
2. 8-HELLFIRE CONFIGURATION
3. WHEEL HEIGHT 20 FEET
4. TRUE AIRSPEED 30 KNOTS



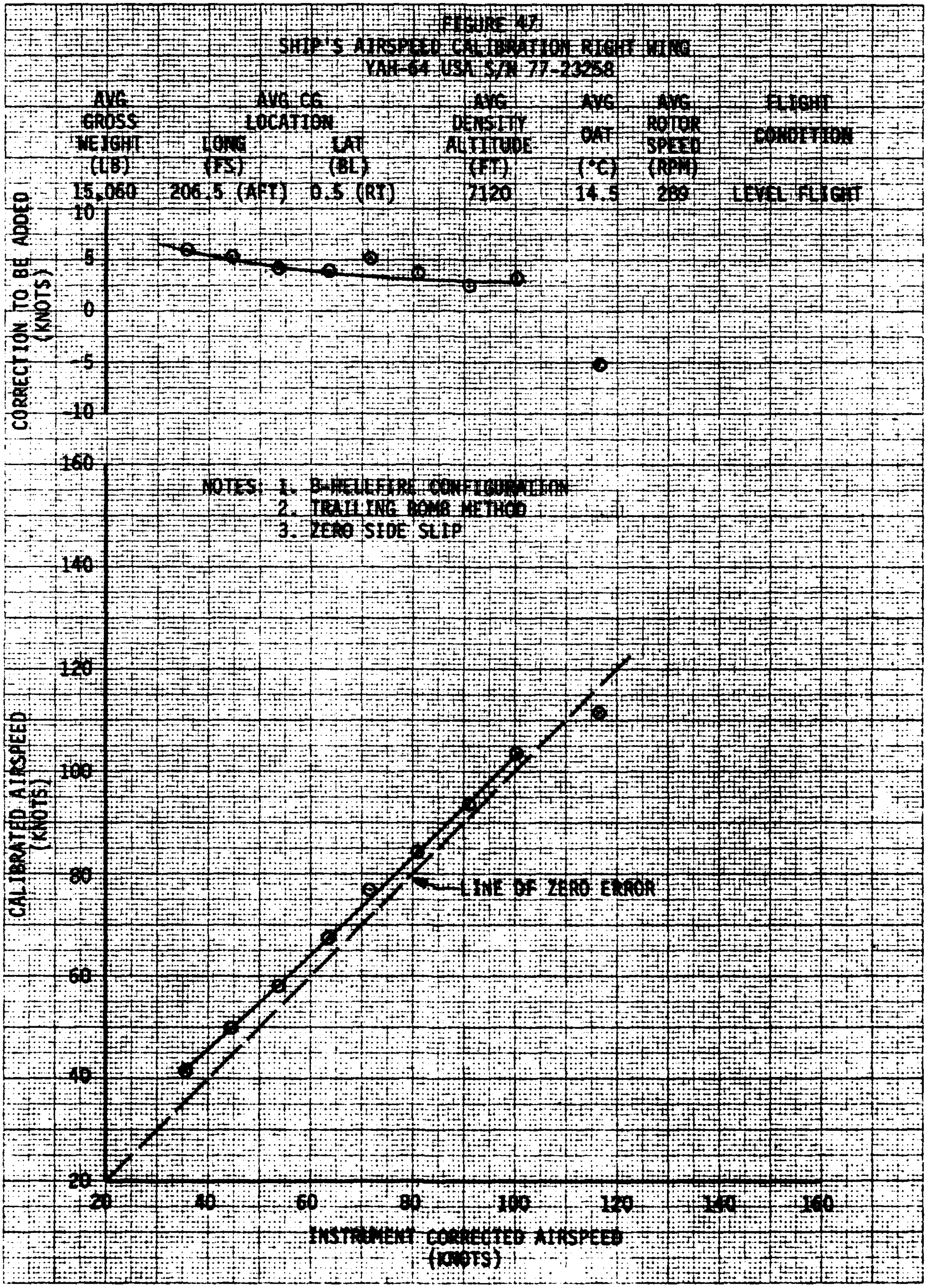
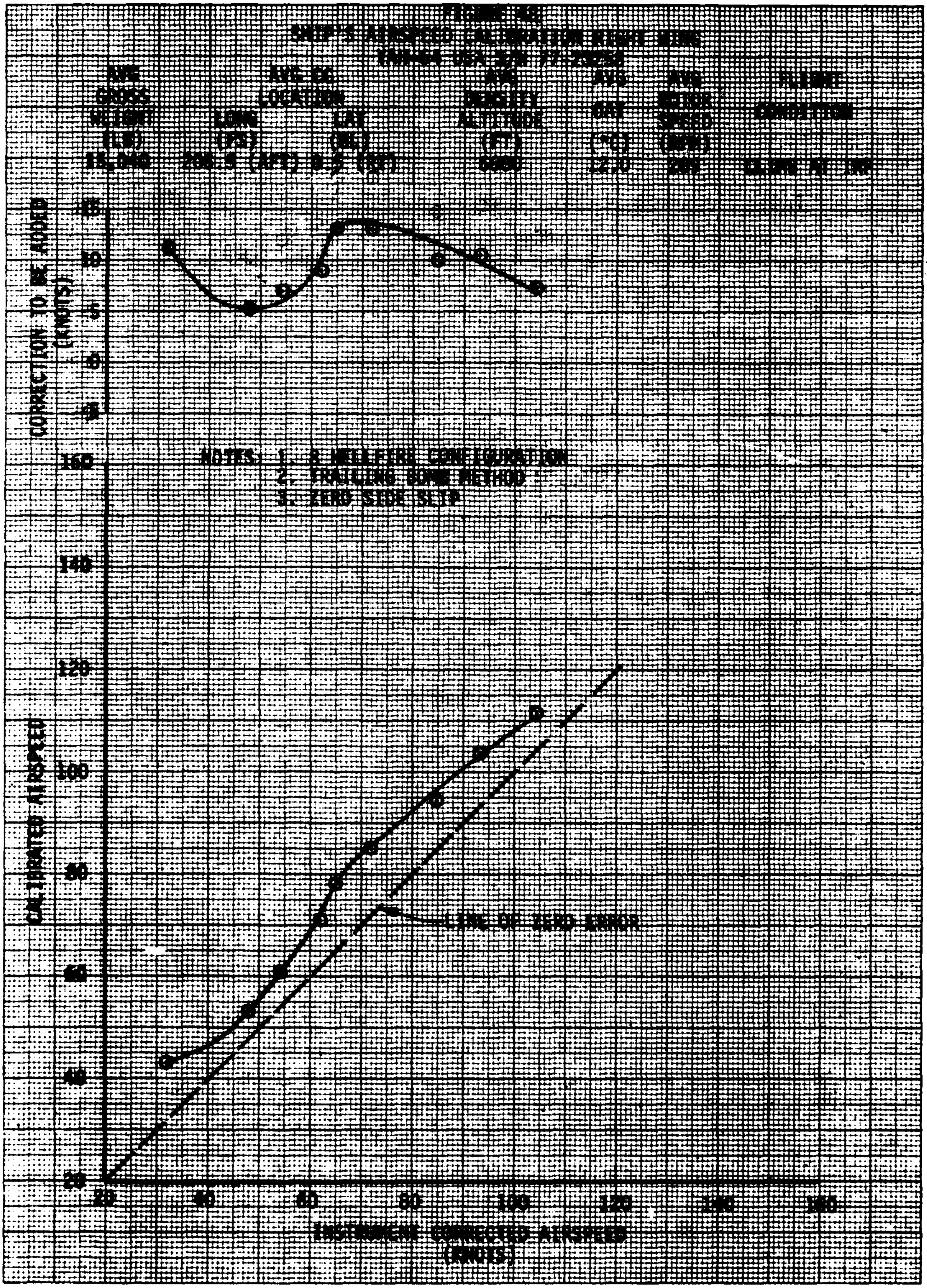


FIGURE 47
SHIP'S AIRSPEED CALIBRATION RIGHT WING
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15,060
206.5 (AFT)
0.5 (RT)
7120
14.5
289
LEVEL FLIGHT
CORRECTION TO BE ADDED (KNOTS)
10
5
0
-5
-10
NOTES: 1. 8-HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD
3. ZERO SIDE SLIP
CALIBRATED AIRSPEED (KNOTS)
160
140
120
100
80
60
40
20
LINE OF ZERO ERROR
20
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)


AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
CORRECTION TO BE ADDED (KNOTS)
15
10
5
0
-5
NOTES: 1. 8 HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD
3. ZERO SIDE SLIP
160
140
120
100
80
60
40
20
CALIBRATED AIRSPEED
LINE OF ZERO ERROR
20
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

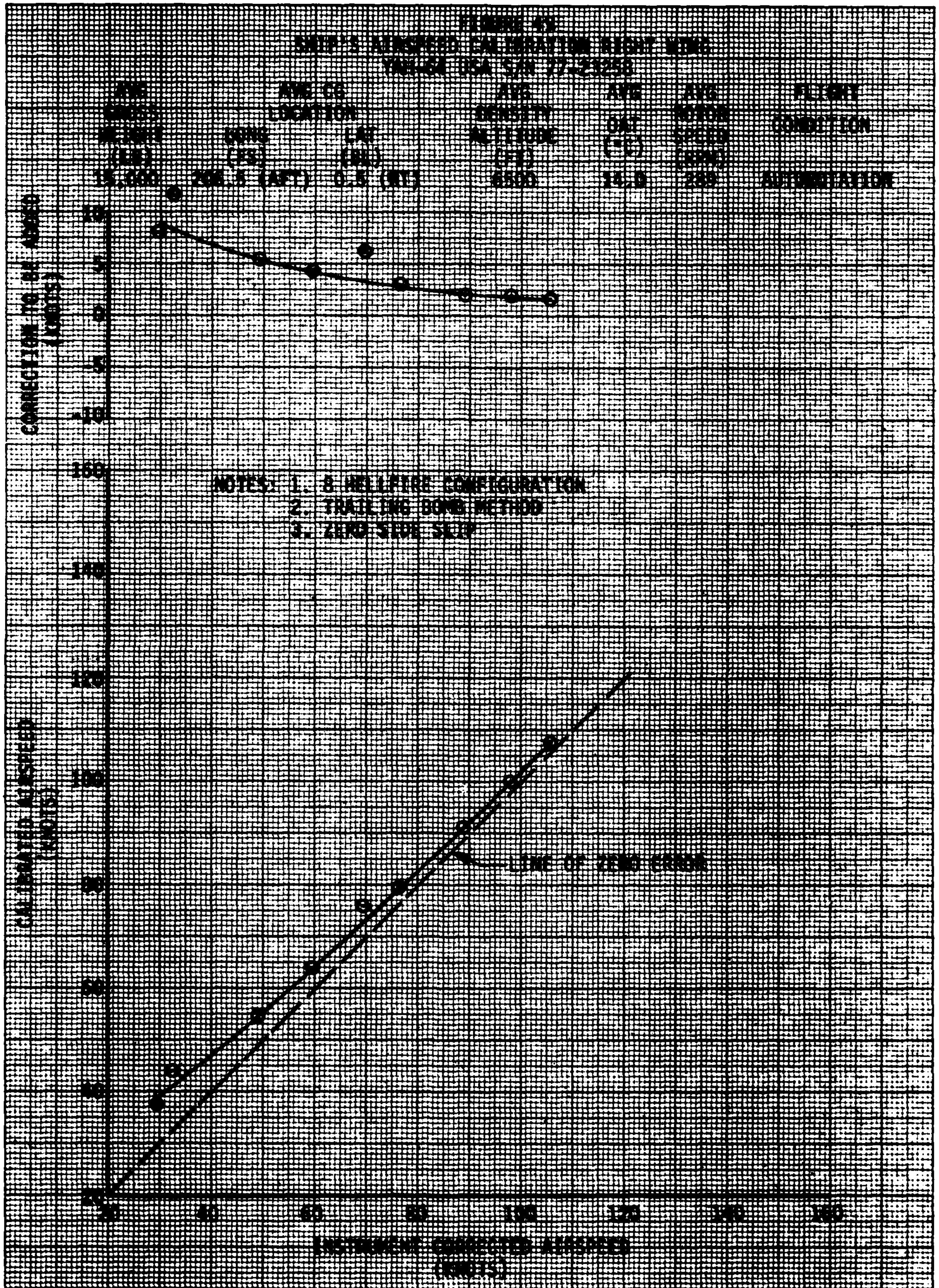
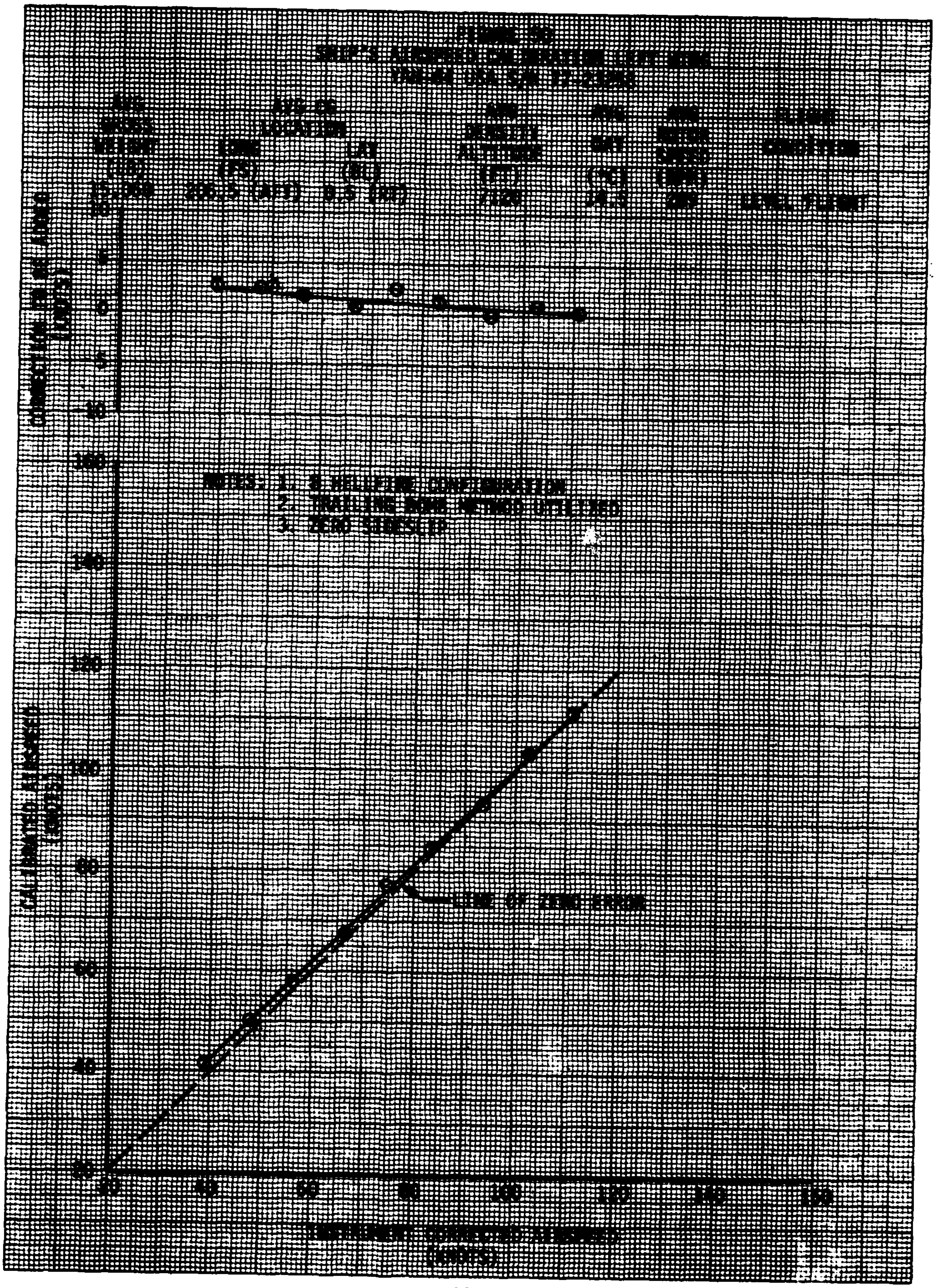
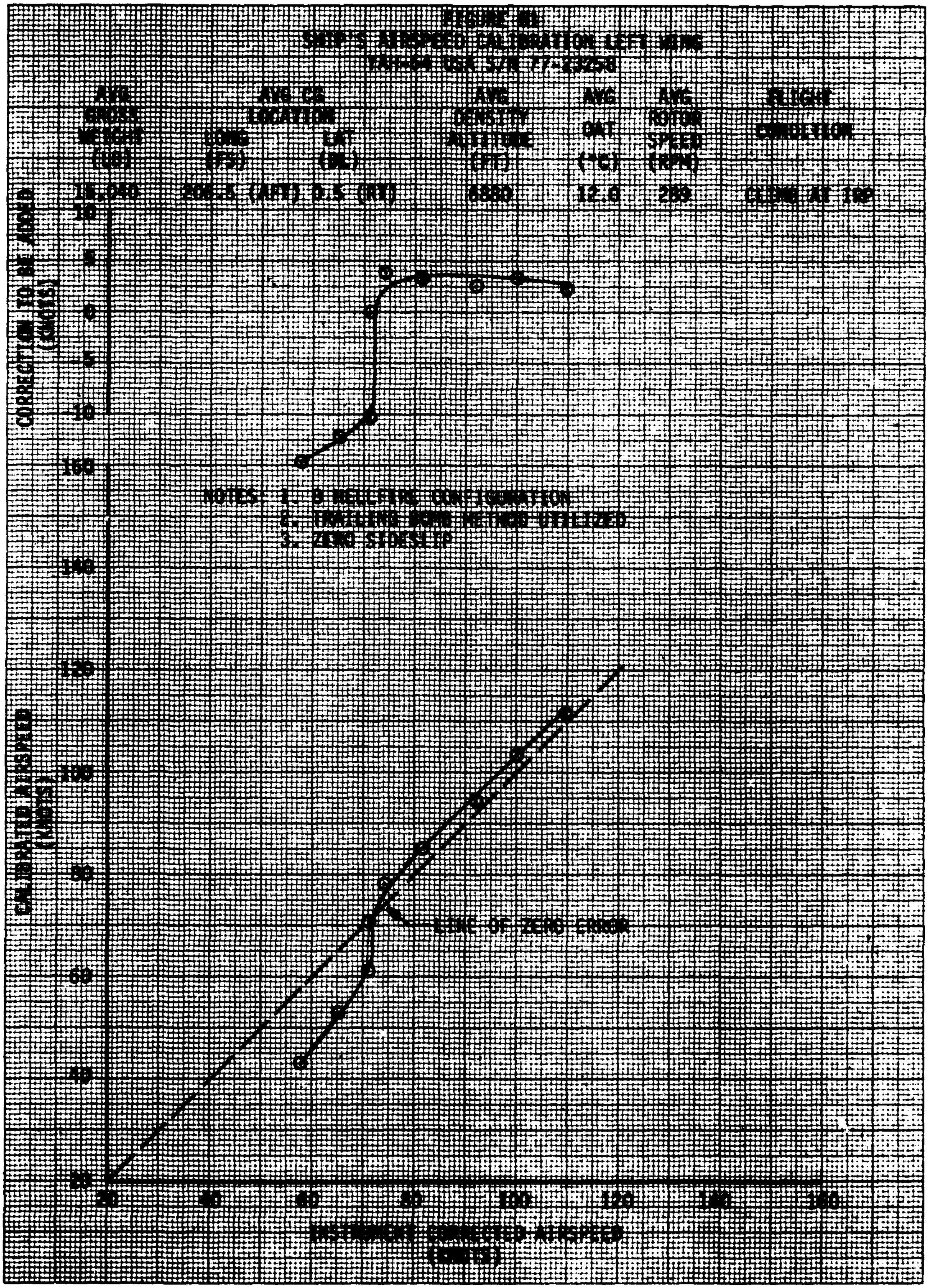
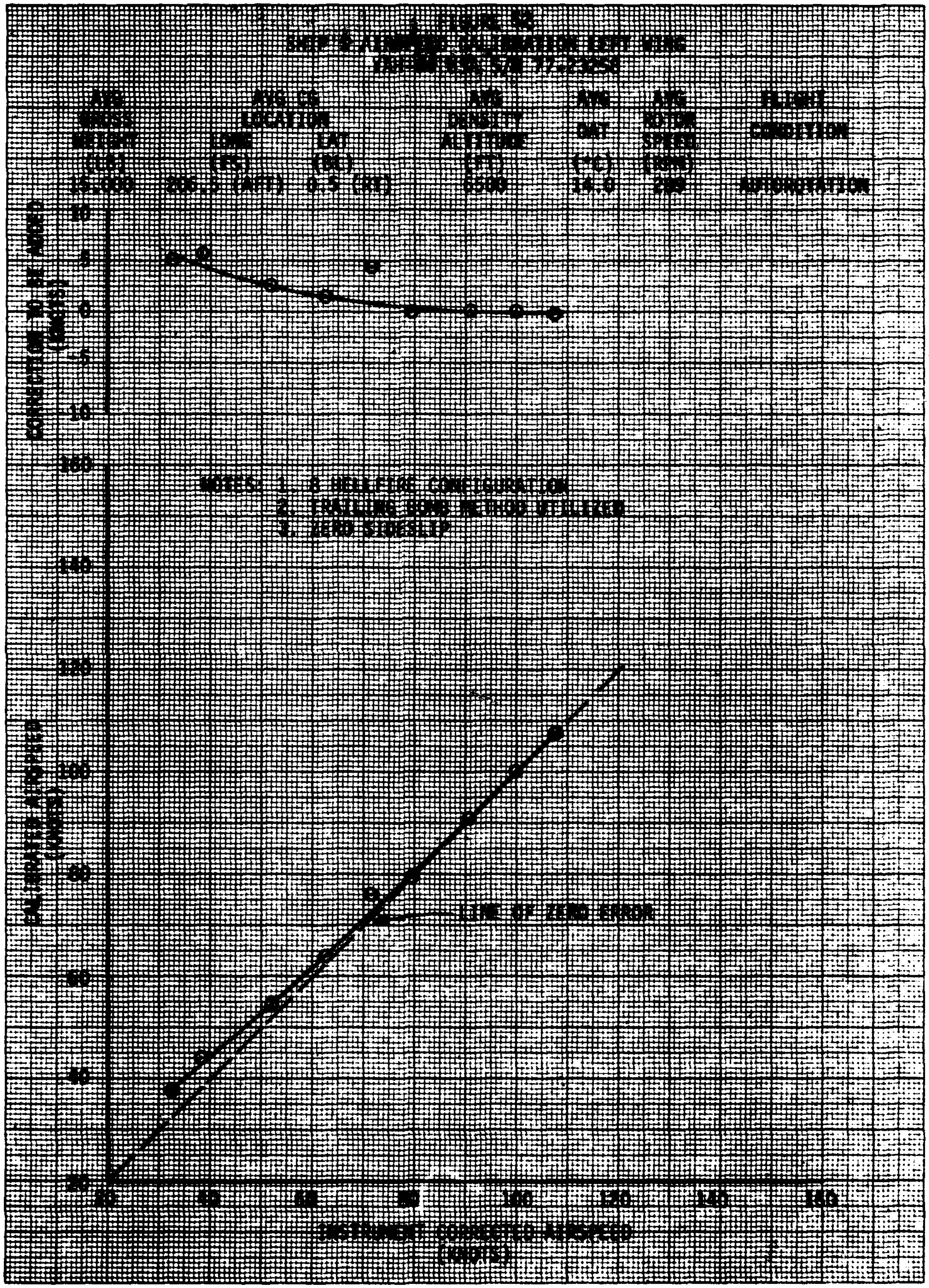
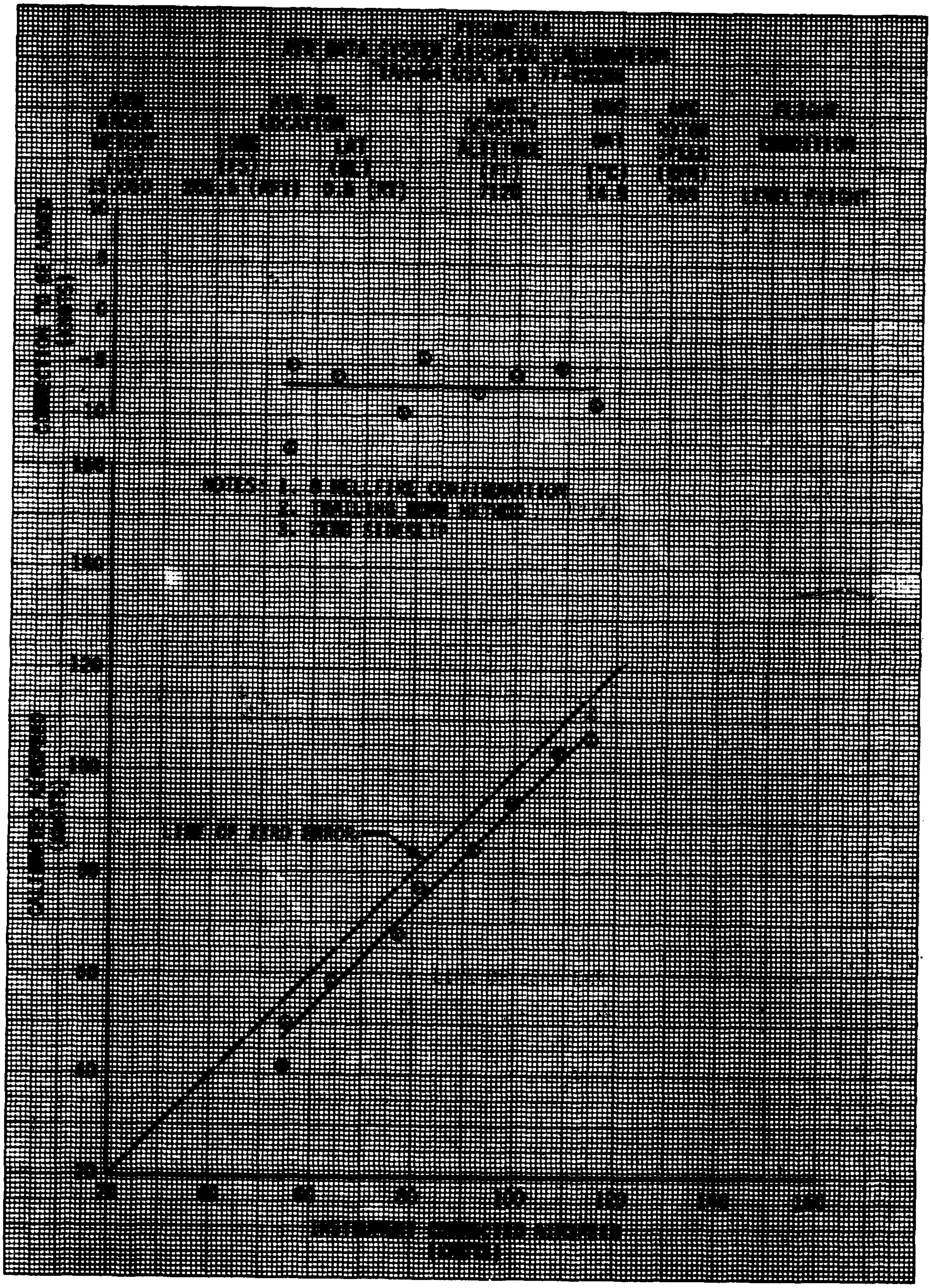


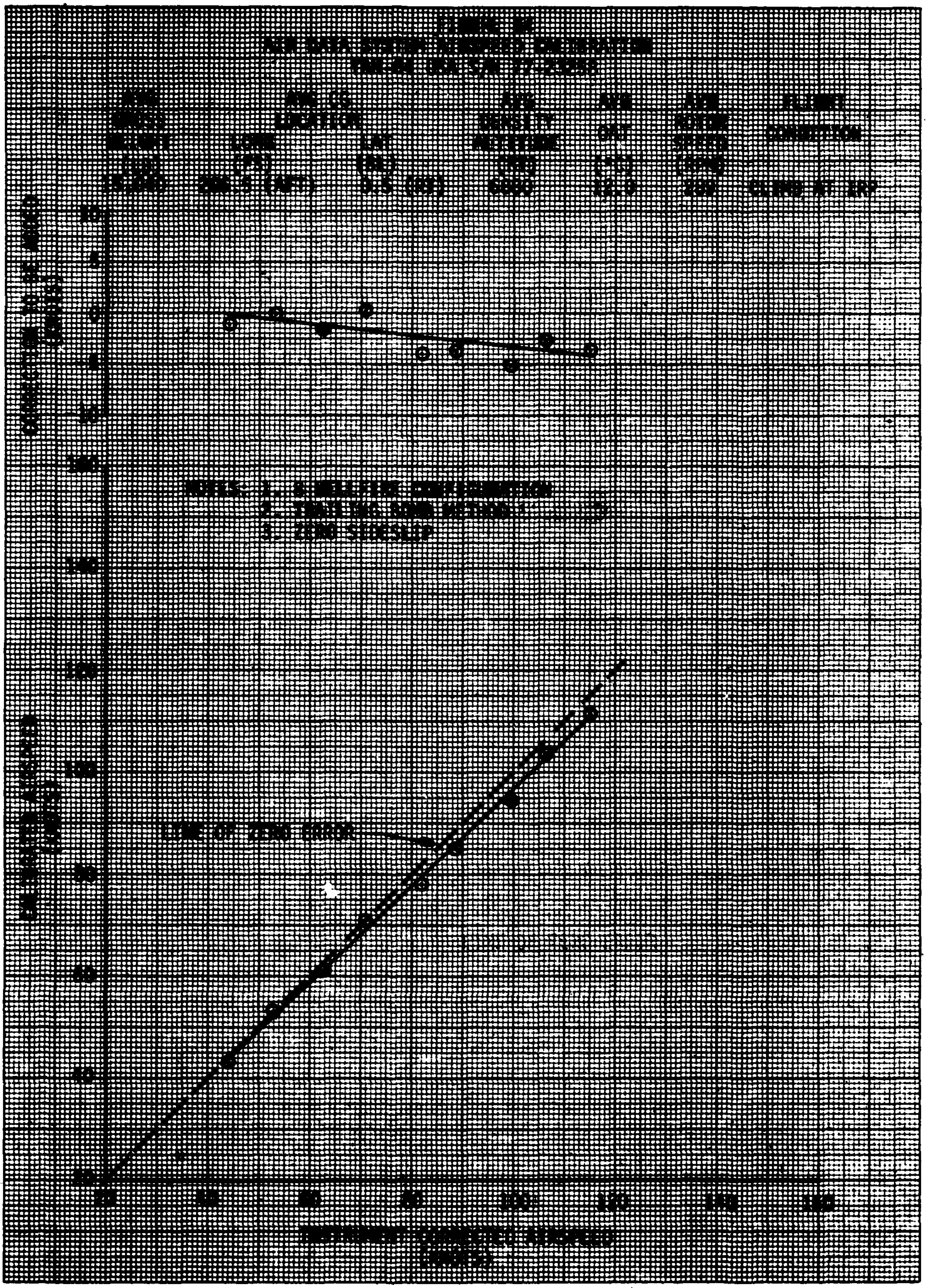
FIGURE 45
SHIP'S AIRSPEED CALIBRATION RIGHT WING
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
15,000
206.5 (AFT)
0.5 (RT)
6500
14.0
289
AUTOROTATION
CORRECTION TO BE ADDED (KNOTS)
NOTES: 1. 8 HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD
3. ZERO SIDE SLIP
LINE OF ZERO ERROR
CALIBRATED AIRSPEED (KNOTS)
INSTRUMENT CORRECTED AIRSPEED (KNOTS)
10
5
0
-5
-10
160
140
120
100
80
60
40
20
40
60
80
100
120
140
160

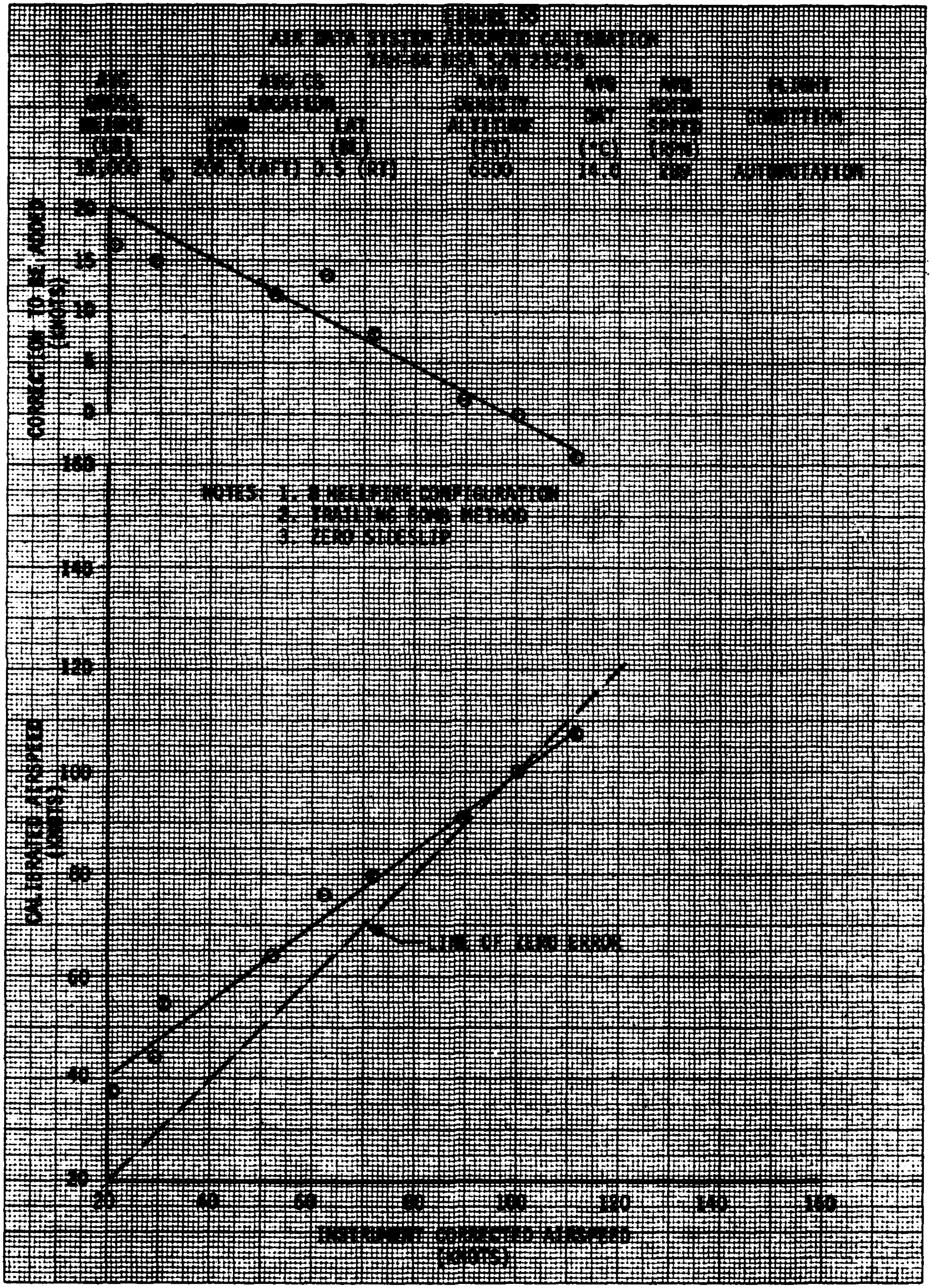

NOTES: 1. 8 HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD UTILIZED
3. ZERO SIDESLIP
20
40
60
80
100
120
140
160

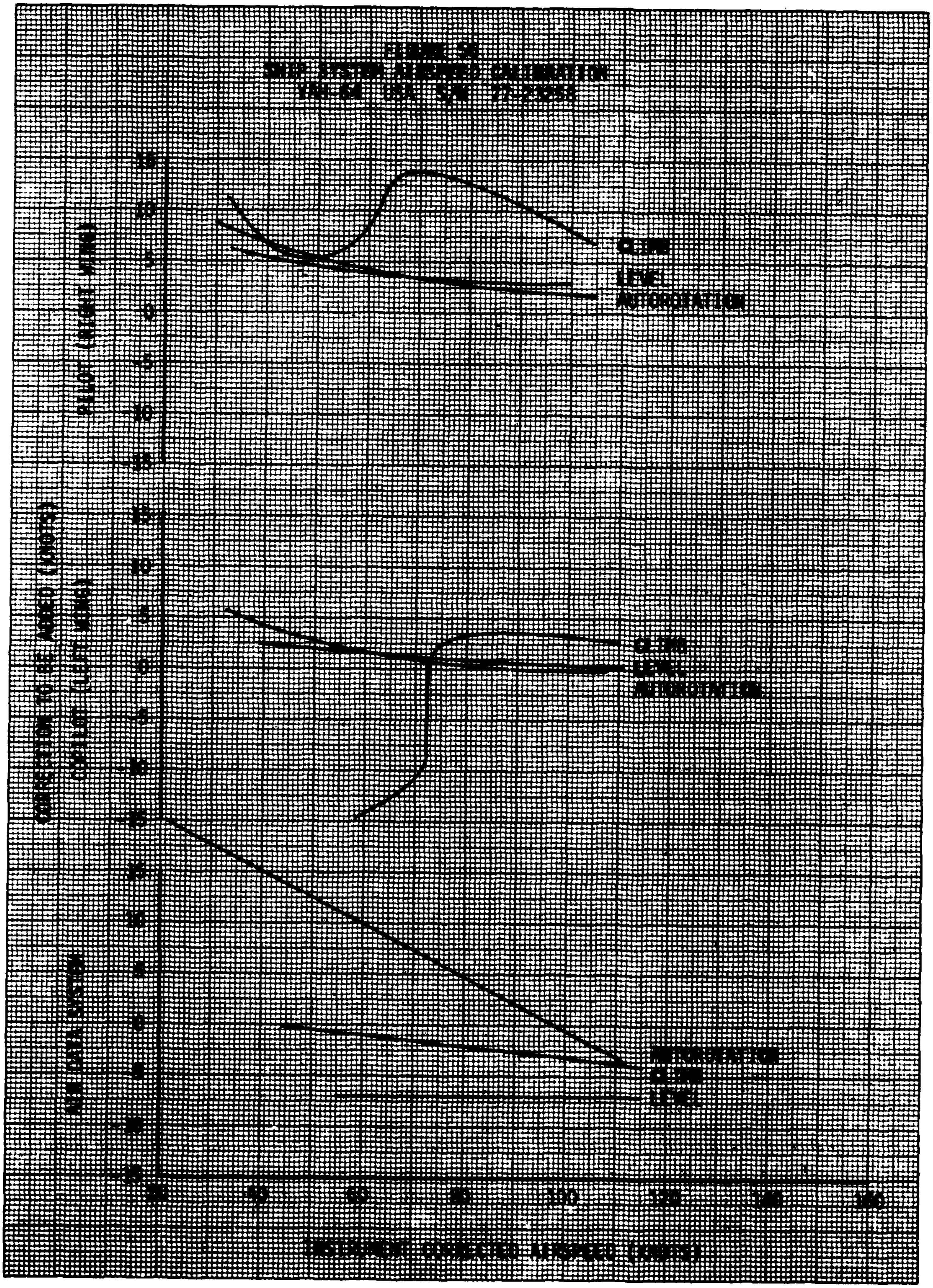

SHIP'S AIRSPEED CALIBRATION LEFT WING
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION
200.5 (AFT) 0.5 (RT)
6880
12.0
289
CORRECTION TO BE ADDED (KNOTS)
NOTES:
2. TRAILING BOMB METHOD UTILIZED
3. ZERO SIDESLIP
CALIBRATED AIRSPEED (KNOTS)
LINE OF ZERO ERROR
INSTRUMENT CORRECTED AIRSPEED (KNOTS)


FLIGHT CONDITION
AUTOROTATION
CORRECTION TO BE ADDED (KNOTS)
10
5
0
-5
-10
NOTES: 1. 8 HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD UTILIZED
3. ZERO SIDESLIP
CALIBRATED AIRSPEED (KNOTS)
160
140
120
100
80
60
40
20
LINE OF ZERO ERROR
20
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)




AVG
GROSS
WEIGHT
AVG CG
LOCATION
LONG
LAT
AVG
DENSITY
ALTITUDE
AVG
OAT
AVG
ROTOR
SPEED
FLIGHT
CONDITION
NOTES:
3. ZERO SIDESLIP
LINE OF ZERO ERROR
CALIBRATED AIRSPEED
INSTRUMENT CORRECTED AIRSPEED
(KNOTS)


AUTOROTATION
CORRECTION TO BE ADDED (KNOTS)
20
15
10
5
0
NOTES: 1. 8 HELLFIRE CONFIGURATION
2. TRAILING BOMB METHOD
3. ZERO SIDESLIP
CALIBRATED AIRSPEED (KNOTS)
160
140
120
100
80
60
40
20
LINE OF ZERO ERROR
20
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)



15
10
5
0
-5
-10
-15
CLIMB
LEVEL
AUTOROTATION
PILOT (RIGHT WING)
CORRECTION TO BE ADDED (KNOTS)
COPILOT (LEFT WING)
AIR DATA SYSTEM
20
40
60
80
100
120
140
160
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

APPENDIX F. EQUIPMENT PERFORMANCE REPORTS

The following EPR's were submitted:

Number		Subject
80-17-2-01	-	Stabilator System *
80-17-2-02	-	Heading and Attitude Reference System
80-17-2-03	-	Air Data Sensor *
80-17-2-04	-	Auxiliary Power Unit
80-17-2-05	-	Environment Control Unit *
80-17-2-06	-	Shaft Driven Compressor *
80-17-2-07	-	Engine Out Warning System
80-17-2-08	-	Digital Automatic Stabilization Equipment *
80-17-2-09	-	Engine TGT Indicator
80-17-2-10	-	Engine Out/Low Rotor Warning Audio Signal *
80-17-2-11	-	Heading and Attitude Reference System
80-17-2-12	-	Engine Nose Gearbox

APPENDIX G. ABBREVIATIONS

a	Speed of sound
A	Main rotor disc area (ft^2)
AAH	Advance Attack Helicopter
AC	Alternating Current
ADS	Air Data System
app	Appendix
APU	Auxiliary Power Unit
AVRADCOM	US Army Aviation Research and Development Command
A&FC	Airworthiness and Flight Characteristics
BL	Butt Line
BUCS	Back-up Control System
C	Celsius
CAS	Command Augmentation System
cg	Center of Gravity
C_L	Centerline
C_P	Coefficient of Power
CPG	Copilot/gunner
C_T	Coefficient of Thrust
DASE	Digital Automatic Stabilization Equipment
DC	Direct Current
deg	Degree
EADI	Electronic Attitude Direction Indicator
ECU	Electrical Control Unit
EDT	Engineer Design Test
ENCU	Environment Control Unit
EPR	Equipment Performance Report
ETP	Experimental Test Procedure
FABS	Forward Avionics Bays
fig.	Figure
fs, FS	Fuselage Station
ft	Feet
HAS	Hover Augmentation System
g	Acceleration of Gravity
GW	Gross Weight
HHI	Hughes Helicopters Incorporated
HMU	Hydromechanical Unit
HQRS	Handling Qualities Rating Scale
Hz	Hertz
IGE	In Ground Effect
IMC	Instrument Meteorological Conditions
in.	Inches
IR	Infrared
IRP	Intermediate Rated Power
KCAS	Knots Calibrated Airspeed
KIAS	Knots Indicated Airspeed
KTAS	Knots True Airspeed
LED	Leading Edge Down
LEU	Leading Edge Up

lb	Pound
LVDT	Linear Variable Displacement Transducer
M_{tip}	Advancing tip mach number
NAMPP	Nautical Air Miles Per Pound of Fuel
NOE	Nap of the Earth
N_G	Gas Generator Speed
N_P	Power turbine speed
N_R	Main rotor speed
OAT	Outside Air Temperature
OGE	Out of Ground Effect
PCM	Pulse Code Modulation
PIO	Pilot Induced Oscillation
PNVS	Pilot Night Vision System
psi	Pounds per Square Inch
Q	Engine output shaft torque
R	Radius (ft)
ref	Reference
RPM	Revolutions Per Minute
SAS	Stability Augmentation System
SCAS	Stability and Control Augmentation System
SHP, shp	Shaft Horsepower
S/N	Serial Number
TADS	Target Acquisition and Designation System
TGT	Turbine Gas Temperature
USAAEFA	US Army Aviation Engineering Flight Activity
V_{cal}	Calibrated Airspeed
VDC	Volts Direct Current
V_H	Maximum Horizontal Velocity
V_{ic}	Airspeed Instrument Error Correction
V_{in}	Indicated Airspeed
VMC	Visual Meteorological Conditions
V_{NE}	Never Exceed Airspeed
V_{pc}	Airspeed Position Error Correction
V_T	True airspeed
VRS	Vibration Rating Scale
WL	Water Line
W_f	Fuel flow rate

Greek and Miscellaneous Symbols

Δ	Incremental change
μ	Advance ratio
ρ	Air density (slugs/ft^3)
σ	Air density ratio
Ω	Main rotor angular velocity (radians/sec)
4/rev	4th harmonic of the main rotor

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